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DEVELOPMENT OF A LOW-COST GRAPHITE REINFORCED COMPOSITE PRIMARY--ETC(U)

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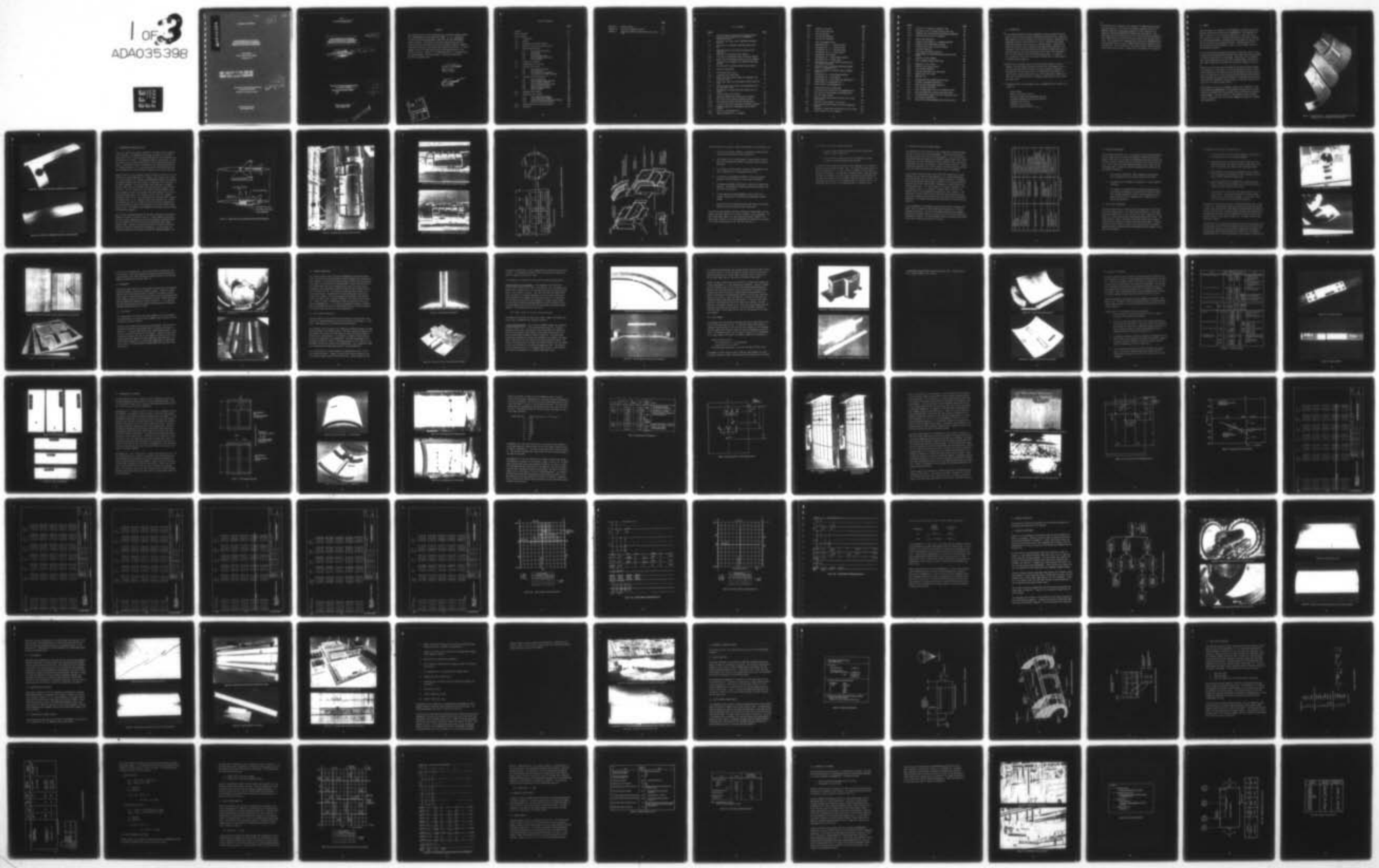

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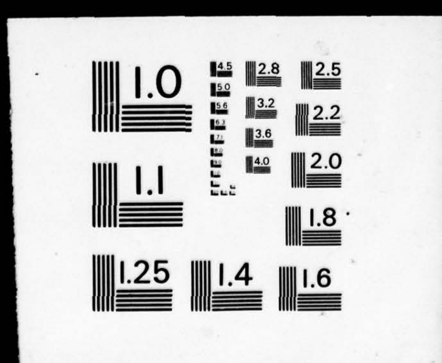

Boeing Report D180-18236-5

DEVELOPMENT OF A LOW-COST GRAPHITE REINFORCED COMPOSITE PRIMARY STRUCTURAL COMPONENT

FINAL REPORT

John H. Laakso and John T. Hoggatt

December, 1976

Prepared Under Naval Air Developmental Center
Contract N62269-74-C-0368

Boeing Aerospace Company
Seattle, Washington 98124

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FOREWORD

This document reports research conducted by the Boeing Aerospace Company under NAVAIRDEVCEN Contract N62269-74-C-0368. Mr. M. S. Rosenfeld Air Vehicle Technology Department, Code 30331, was the NAVAIRDEVCEN Project Engineer. The technical activities reported were under the direction of the undersigned of the Boeing Aerospace Company's Advanced Structures R&D Group in the Research and Engineering Division. Mr. John T. Hoggatt, also in the same group, directed the reported material and process activities. Messers. H. A. Bjornestad and J. W. Straayer served as program managers during the program. The program duration was May 1974 to September 1976.

John H. Laakso
Technical Leader

J. L. Arnquist
Program Manager

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1.0 INTRODUCTION

In two preceeding NAVAIR programs (References 1 and 2), materials and processes studies indicated graphite reinforced polysulfone (Gr/Ps) composites have potential cost saving benefits for aircraft structure. Because of the thermoplastic material characteristics of Gr/Ps, laminates can be made, stored and then post-formed anytime to a desired contour. Chemistry and shelf-life problems, such as associated with epoxy-based composites, are eliminated. The simplified handling, post-forming by simple application of heat and pressure, and recycling possibilities of Gr/Ps offer good potential for cost saving in aircraft structure applications. In addition, Gr/Ps composites should offer structural efficiency that is representative of graphite/epoxy composites.

Based on the encouraging results of the preceeding programs, the Navy funded a joint Boeing/Navy program to take the next developmental step to build and evaluate some Gr/Ps primary aircraft structural components. The objectives of the program were to (1) demonstrate the potential for cost savings attainable through the use of the thermoplastic composite concept, and (2) identify benefits to a weapon system offered by Gr/Ps composite application.

The Boeing portion of the program, which is summarized in this report, consisted of:

Design Studies,
Material and Process Studies,
Structural Element and Subcomponent Testing,
Prototype Component Fabrication and Delivery,
Component Test Planning,
Comparative Cost Analysis,
Post-test Component Analysis.

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The prototype Gr/Ps components were designed, for demonstration purposes and low program cost, to replace the aluminum centerbody skins of the XBQM-34E Supersonic Firebee Target drone, ("X" prefix indicates pre-production configuration), built by Teledyne Ryan Aeronautical. This application area was chosen because of its real-world design complexities and acceptability for structural airframe testing within funding limitations. The prototype Gr/Ps components were delivered to NAVAIRDEVCEN who installed the parts on a surplus XBQM-34E and performed laboratory structural testing. At the conclusion of the ground testing, Boeing performed a proof-test analysis which is summarized in an appendix of this report.

2.0 SUMMARY

Three prototype Gr/Ps components for the XBQM-34E centerbody section were designed, fabricated and delivered to NAVAIRDEVCEN. The components are shown in Figures 2-1 to 2-3 and are identified as a right side skin, a left side skin and a door. Structural analyses certified their integrity for the BQM-34E (production configuration) design conditions based on the results of element and subcomponent testing. Including penalties for retro-fitting, the Gr/Ps components offer a 5 percent weight saveing in the XBQM-34E application.

As part of the design development activities, a preliminary material specification for Gr/Ps prepreg was prepared, and a novel sheet stock technique for laminate fabrication was established. Fabrication of the component parts were accomplished using stock Gr/Ps laminate sheets, low-cost tooling, simple thermo-forming methods and conventional inspection methods.

Comparative cost analyses were performed on preliminary designs prepared for a hypothetical large-scale aircraft fuselage panel using (1) conventional aluminum and (2) graphite-reinforced thermoplastic design concepts. Based on extrapolations of the processes used to fabricate the delivered Gr/Ps components and additional assumptions, a 20 percent cost saving was offered by the thermoplastic design concept in a 100-unit production run along with a 16 percent weight saving.

Ground testing of the Gr/Ps components installed on a XBQM-34E airframe was performed by NAVAIRDEVCEN. The components sustained the limit load condition associated with chute recovery (the critical design condition) without damage. Post-test analysis of the measured strains indicates that the load capabilities of the Gr/Ps components exceed their ultimate load requirements.

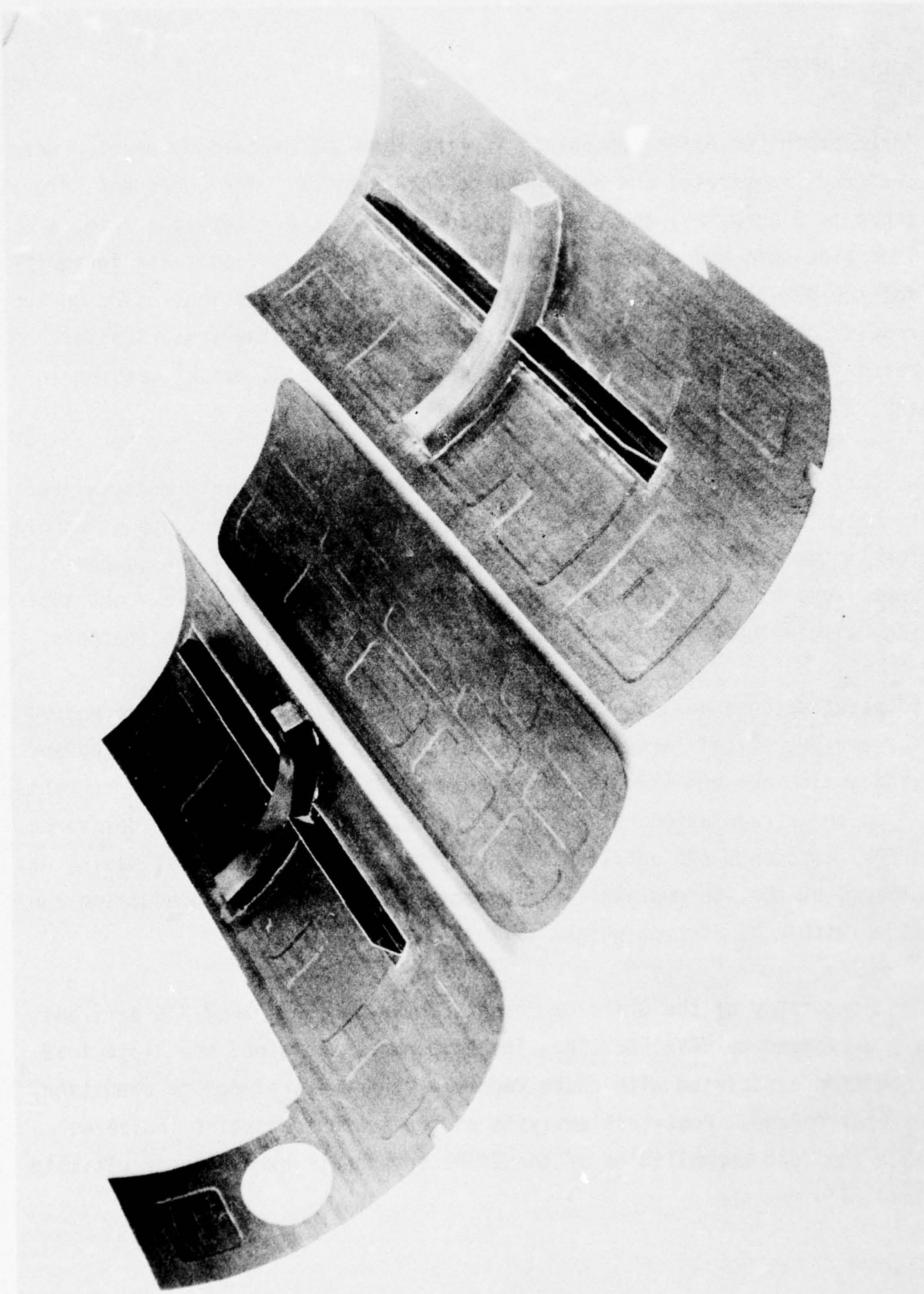

Figure 2-1. Delivered Graphite – Reinforced Polysurfone Thermoplastic (Gr/Ps) Components for the XBQM-34E Centerbody Section

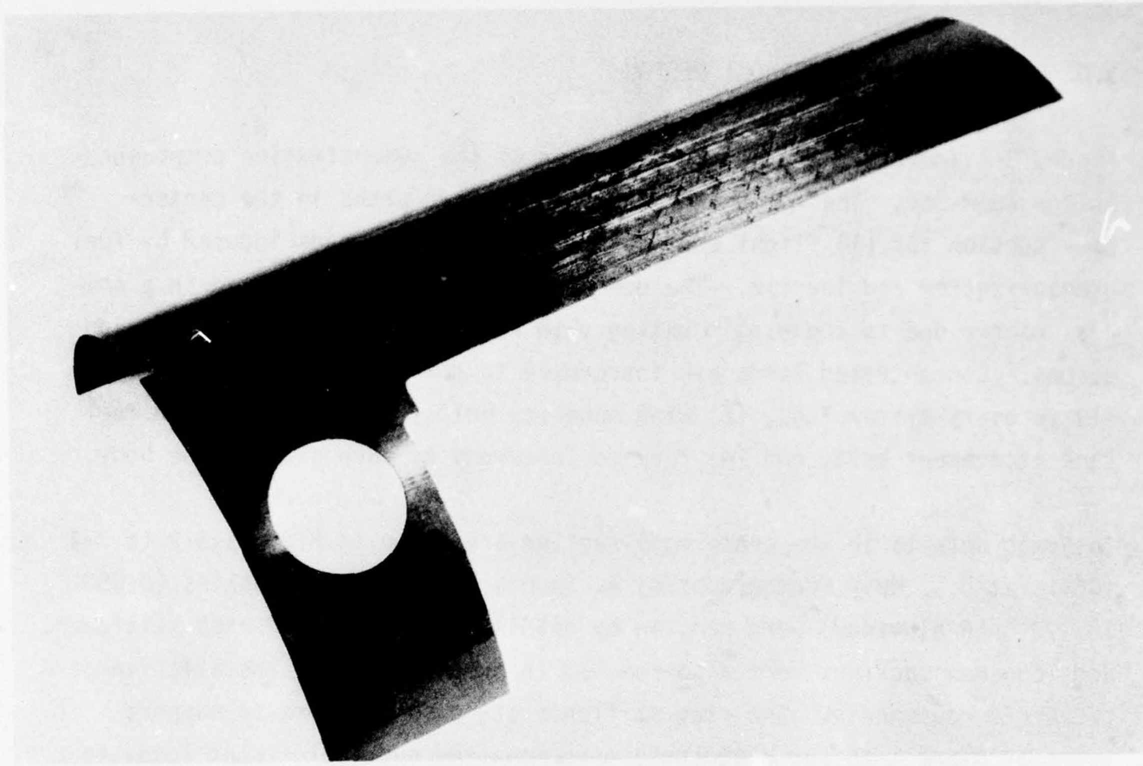

Figure 2-2. Gr/Ps Skin for Right Side of XBQM-34E Centerbody Section

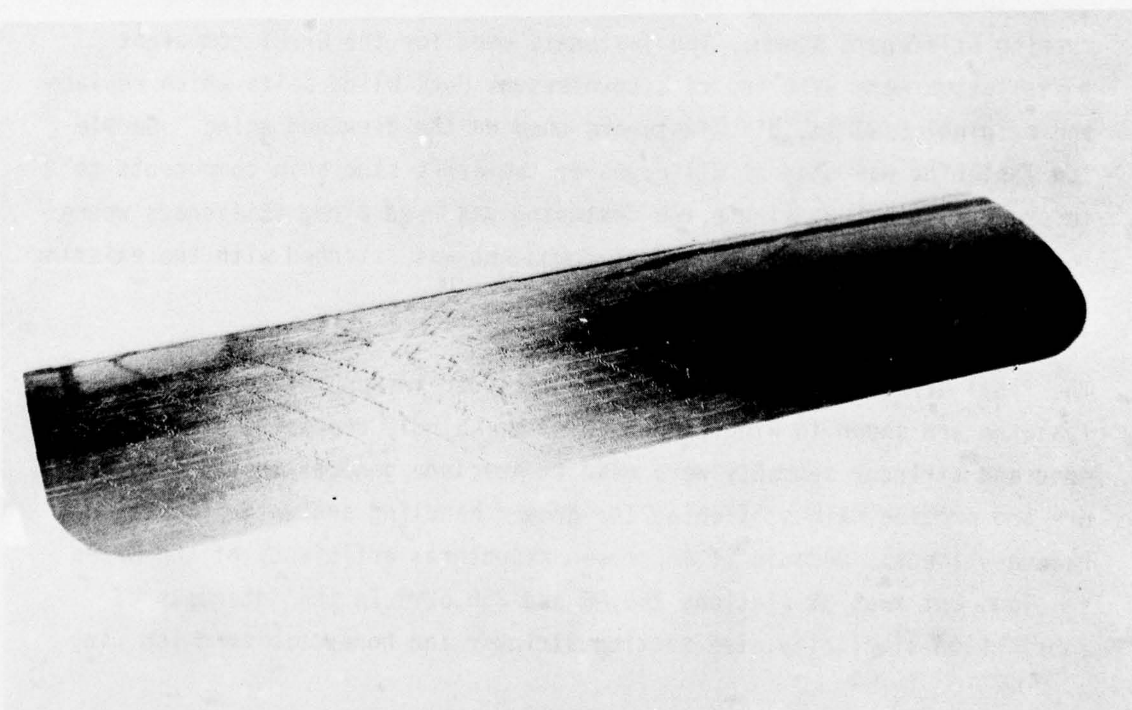

Figure 2-3. Gr/Ps Panel for XBQM-34E Centerbody Lower Access Door

3.0 DEMONSTRATION COMPONENT DESIGN

Figure 3-1 indicates the general location of the demonstration components on the XBQM-34E. The components are primary load paths in the centerbody section for (1) flight and ground loads and (2) loads induced by fuel pressurization and inertia. The component loads are distributed in a complex manner due to the wing mounting skin slot and discontinuous body longerons. Concentrated loads are introduced to the centerbody section via (1) recovery system lugs, (2) wing mounting bolts, (3) external drop fuel tank attachment bolt, and (4) forward longerons at each side of the body.

Interior details in the centerbody section are shown in Figures 3-2 to 3-4 (Official U.S. Navy Photographs by A. Shanks). The original skins (0.050 in. 7075-T6 aluminum) were removed by NAVAIRDEVCEN; selected ring stiffener and longeron sections were also removed in preparation for installation of the Gr/Ps components. The ring stiffener stubs shown serve to support interior pipe lines and distribute wing mounting and fuel system loads to the skins. The skins fasten to three machined aluminum bulkheads, wing mount longerons, recovery lug fittings, door jamb longerons and to the remaining stiffeners shown. The fasteners used for the Gr/Ps component installation were 3/16 in. dia. countersunk Huck blind bolts which replace the original 5/32 in. dia. fasteners used on the aluminum skins. Double row fastening was used at all edges of the Gr/Ps side skin components to insure fuel sealing; single row fastening was used along stiffeners where loads are low. The removable door component was attached with the existing 118 countersunk fasteners.

The final Gr/Ps component designs are included in Appendix A and the basic features are shown in Figures 3-5 and 3-6. Single composite ring stiffener and stringer segments were used to preclude general instability failure and provide skin stiffening for ground handling and water impact recovery loads. Because of increased structural efficiency of the Gr/Ps sections cut away at stations 250.06 and 266.62. In the interests of fabrication simplicity, tee-section stringer and honeycomb sandwich ring

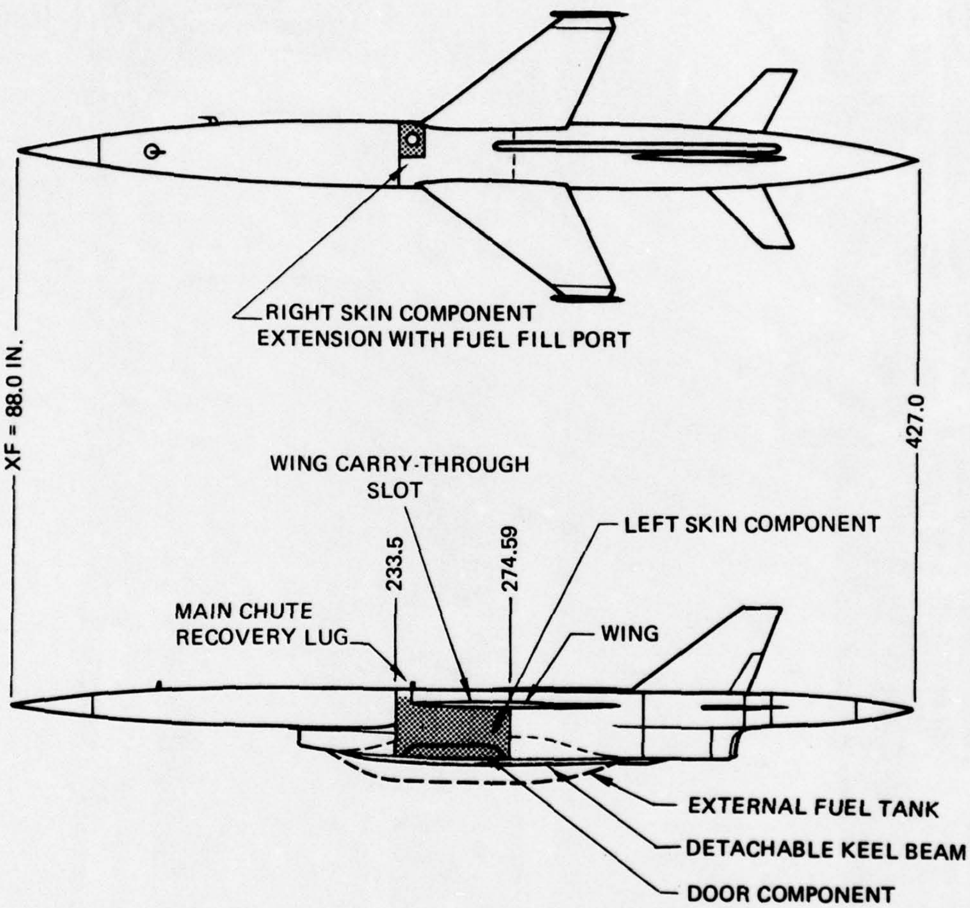

Figure 3-1: XBQM–34E Profiles and Selected Prototype Gr/Ps Components

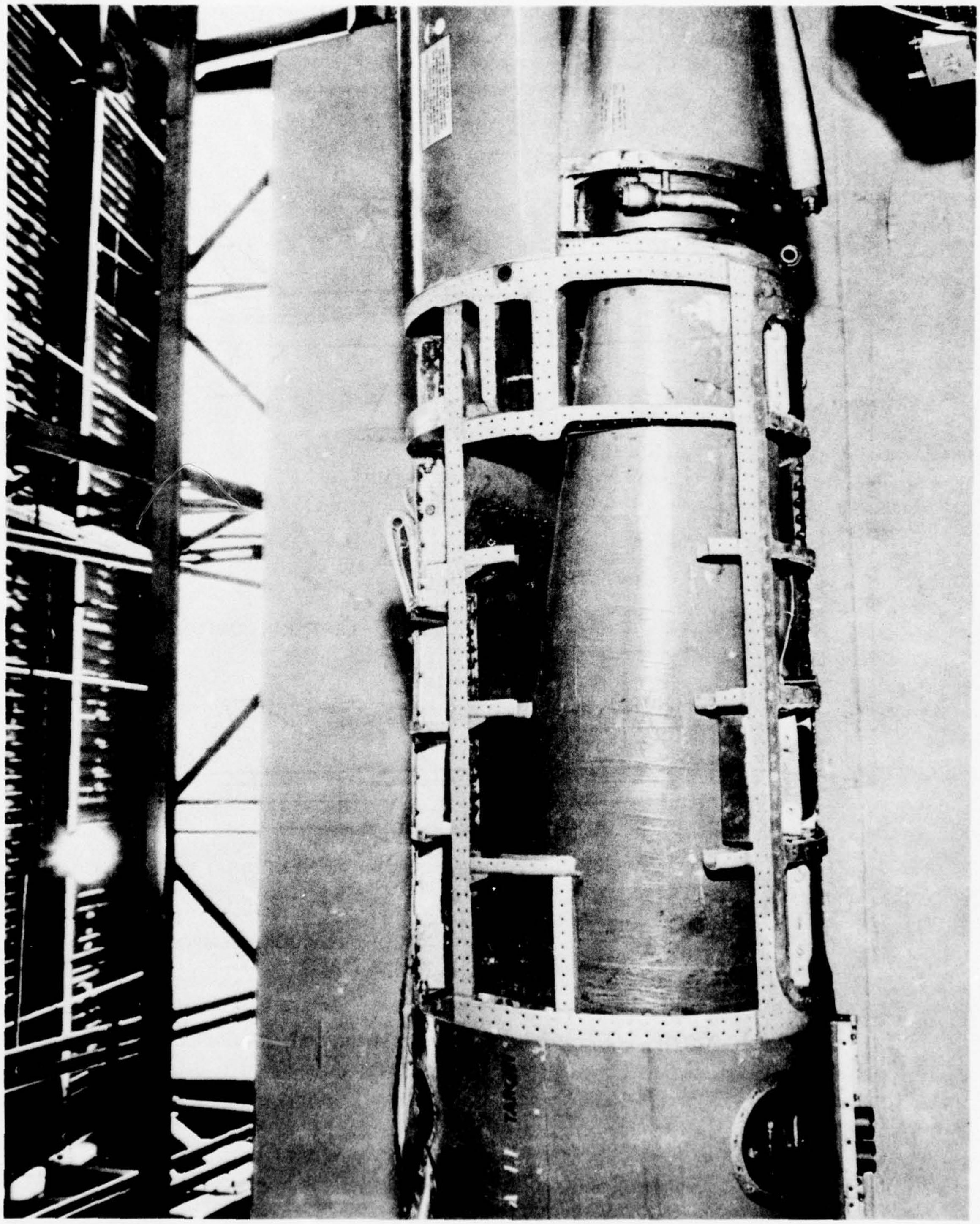

Figure 3-2. XBQM34E With Centerbody Skins Removed

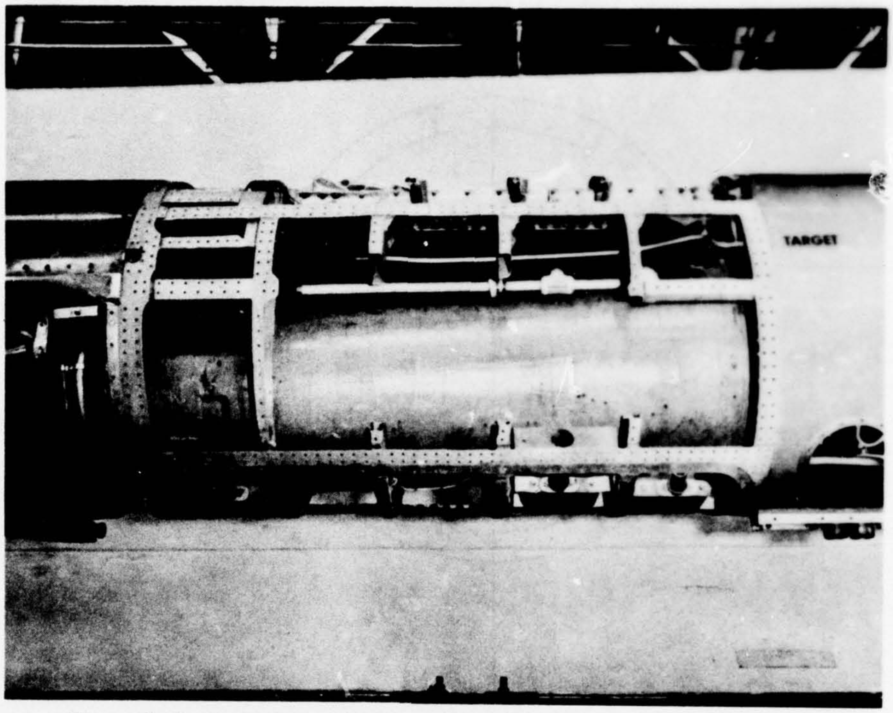

Figure 3-3. Left Side of Centerbody Section With Skins Removed

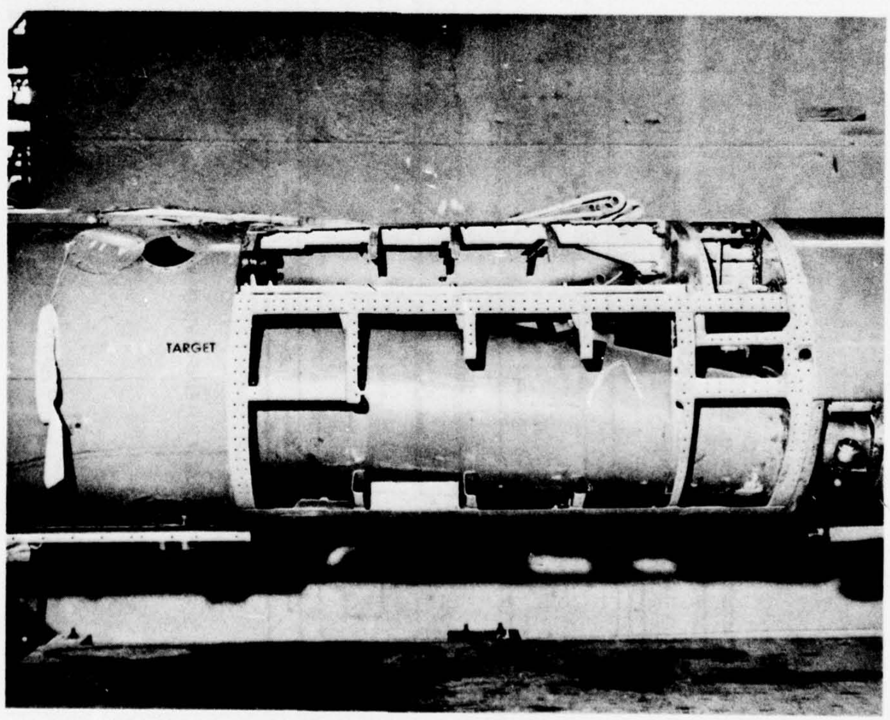

Figure 3-4. Right Side of Centerbody Section With Skins Removed

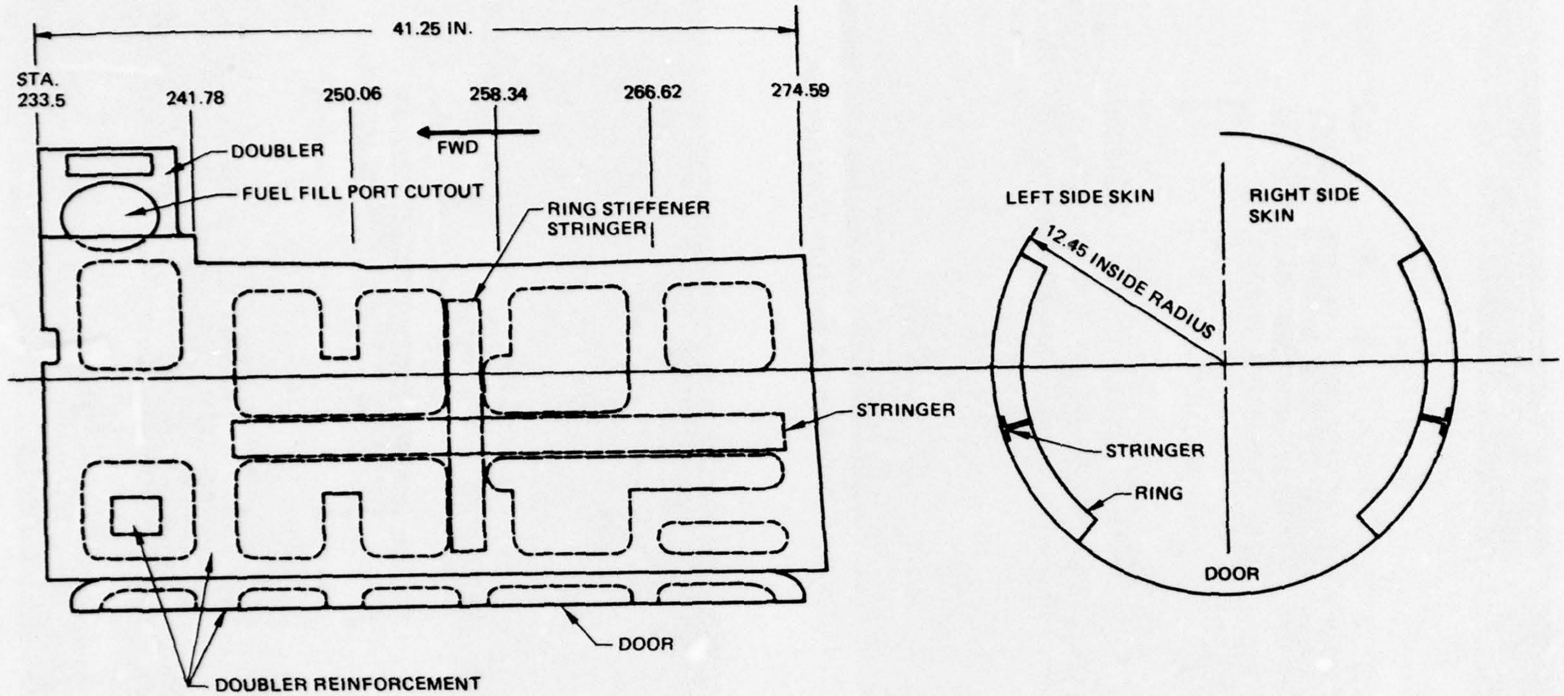

Figure 3-5: Prototype Gr/PS Components for XBQM-34E Centerbody Section

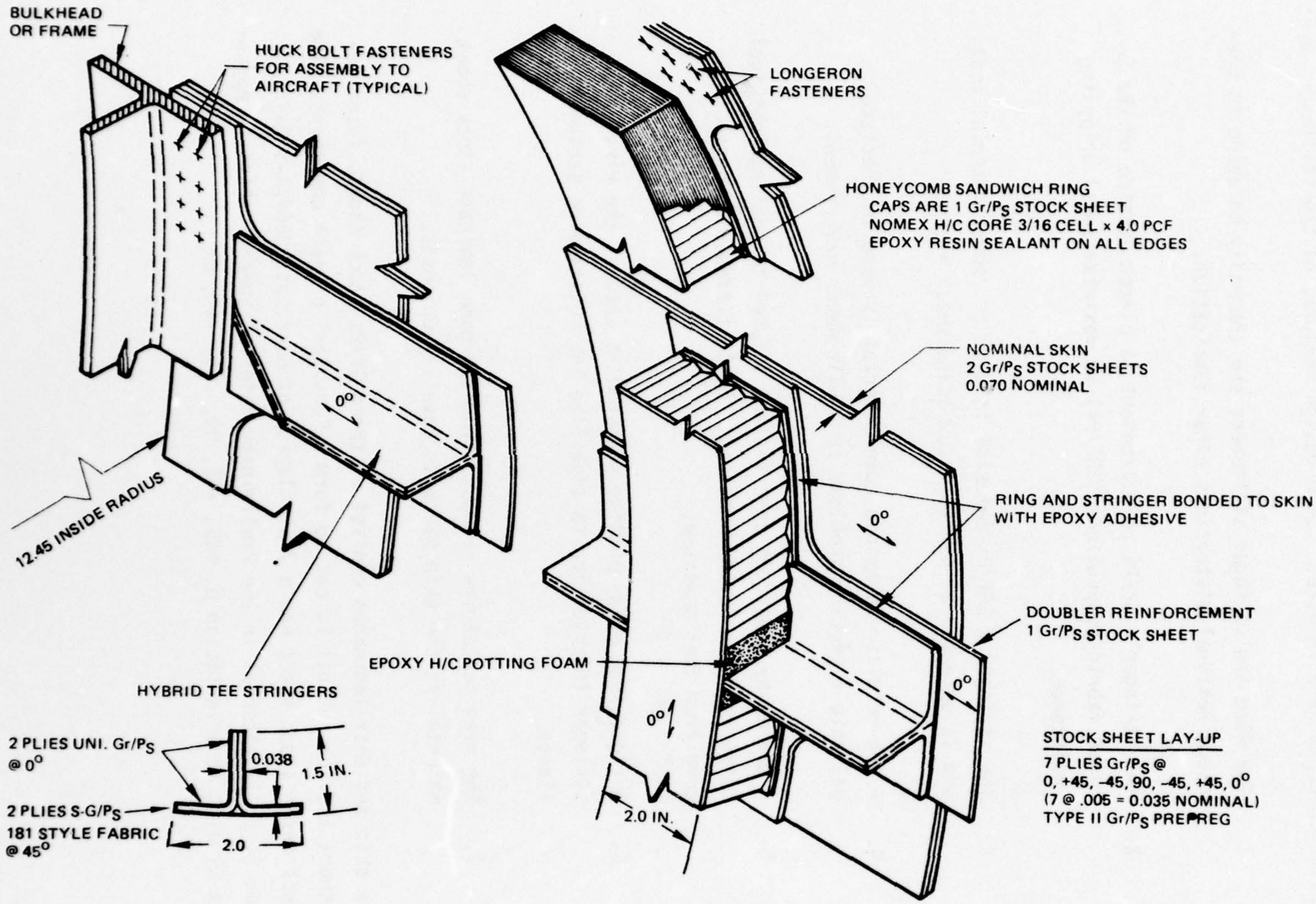

Figure 3-6: Prototype Gr/Ps Component Design Concept Details

design concepts were selected. Some design features of the stiffeners are:

1. The ring and stringer stiffeners are thermally balanced to preclude residual distortions after fabrication.

2. The stringer's Gr/Ps reinforcement is placed outside of the S-glass fabric/polysulfone (SGF/Ps) to maximize local crippling resistance.

3. The stringer's SGF/Ps material is used to reduce material cost and is oriented at 45° to accommodate shear flow.

4. A non-metallic honeycomb core material is used to preclude galvanic corrosion problems in a salt water environment.

5. Rigid epoxy honeycomb core sealant is used to seal against fuel moisture. The sealant has sufficient stiffness to protect the core from fuel pressure.

6. A rigid foam epoxy potting compound is used at the ring-stringer intersection to stabilize the outstanding stringer flange.

7. The ring cap strips are made from the same laminate stock sheet material as the skin parts to save lay-up costs.

The skin and door laminates consist of three Gr/Ps stock sheets fused together; the inner sheet is cut to form a "doubler" on each component. The basic Gr/Ps stock sheet has a 7-ply lay-up of Hercules' 3004/A-S 3-in. tape (Type II material in the Preliminary Material Specification in Appendix C) with an orientation 0, +45, -45, 90, -45, +45, 0°.

This laminate lay-up was selected to provide:

1. A nearly isotopic balanced laminate stock sheet having good bearing strength properties.

2. A thin flexible laminate for ease of preforming to a tight cylindrical curvature without fiber buckling.

The doubler sheet reinforces the skin and door components in all areas of fastener holes and along stiffener lines. The resulting nominal panels are two stock sheets thick (2 x 0.035 = 0.070 in. nominal) and are capable of post-buckling behavior. The placement of doubler material under the bonded stiffeners is believed to restrict pre-or post-buckling displacements to the unreinforced panel areas so stiffener "peeling" should not occur. The resulting skin thickness at the component edges is nominally 0.105 in. so knife edges do not exist in the countersunk 3/16 in. fastener holes.

4.0 MATERIAL SELECTION AND QUALIFICATION

The thermoplastic resin selected for this program was P-1700 polysulfone, a product of the Union Carbide Corporation. The selection was based on previous experience with the material (References 1 to 3) and structural and environmental requirements of the BQM-34E Firebee Target Drone. Other commercially available thermoplastic resins systems may have also met the requirements but were not available in graphite prepreg tape form. Boeing had working experience with P-1700 on glass, Kevlar and graphite reinforcements which offered flexibility to the design concept development.

The principal reinforcing fiber used in the body component was A-S type graphite with the material being supplied as 3 in. wide preimpregnated tape from Hercules, Inc. with product designation 3004/A-S. The material was purchased and qualified as Type II material to the Boeing preliminary material specification D180-18236-4, "Thermoplastic Graphite Reinforced Preimpregnated Tape." This specification is enclosed in Appendix C. As an alternate source, U. S. Polymerics' prepreg, 3023, which has T-300 graphite fiber as reinforcement, was qualified as Type I material. The 3004 system provided slightly better properties and uniformity than the 3023 prepreg system and was the basis for its selection for the prototype components. The system requirements which both materials met are given in Figure 4-1.

In the development of the component design concept, fabric prepregs with glass, Kevlar and graphite reinforcements were evaluated in a 181 style 8 harness-satin weave. The Kevlar and graphite fabrics were preimpregnated with P-1700 polysulfone by the Hexcel Corp. and the glass fabric was impregnated by Boeing. The design concept finally selected utilizes SGF/Ps as the fabric prepreg form because of low cost.

PROPERTY	BQM-34E REQUIREMENTS	QUALIFICATION
MECHANICAL PROPERTIES	D180-18236-4 MINIMUM VALUES	MATERIAL QUALIFIED TO D180-18236-4
JP-4 FUEL COMPATIBILITY	CONTINUOUS EXPOSURE AT +77°F	NO EFFECTS AFTER 28 DAY EXPOSURE
HYDRAULIC FLUID EXPOSURE MIL-H-5606	CASUAL EXPOSURE	NO EFFECT AFTER 28 DAY EXPOSURE
SALT-WATER EXPOSURE	WITHSTAND SALT SPRAY AND WATER IMMERSION	NO EFFECT AFTER 200 DAY EXPOSURE IN SALT SPRAY NO EFFECT AFTER 60 DAY EXPOSURE IN SALT WATER
SEALANT COMPATIBILITY	PROCESS AND CHEMICAL COMPATIBILITY	NO EFFECT - NO SOFTENING OR DETERIORATION OF COMPOSITE RESULTING FROM APPLICATION OF POLYSULIFIDE TANK SEALANT
PAINT COATING	PROCESS AND CHEMICAL COMPATIBILITY PER MIL-F-18204	GR/PS PANEL SUCCESSFULLY PRIMED AND PAINTED PER MIL SPECIFICATIONS
FIRE RESISTANCE	NON-BURNING	QUALIFIED PER ASTM D635-63
SYNTHETIC LUBRICANT MIL-L-7807	CASUAL EXPOSURE	NO EFFECT AFTER 28 DAY EXPOSURE
WEATHER EXPOSURE	CONTINUOUS EXPOSURE	NO EFFECT PER ASTM D1499

Figure 4-1: Material Requirements

5.0 FABRICATION PROCESSES

The fabrication processes used in this program for the reinforced thermoplastic composites were selected and designed to emphasize minimum production costs, while at the same time demonstrate application feasibility. Funding limitations restricted the use of production techniques and tooling. But to insure that the technology transfer could be made from the prototype stage to production, several requirements were imposed on the developmental processes:

1. All processing conditions, (time, temperature and pressure) must be compatible with available production facilities.

2. Procedures must be amenable to automation, or rapid production techniques.

3. All aspects of the production cycle must be considered even if not actually implemented in the feasibility study and costing. This includes raw material control, quality assurance, scrappage, repairability and inspectability in addition to the processing operations.

5.1 STOCK SHEET FABRICATION

The basic approach taken to fabrication of skin parts was to start with basic sheet stock material. The stock sheet consisted of a 7-ply layup having an orientation of 0, +45, -45, 90, -45, +45, 0° made from .005-in. thick, 3-in. wide unidirectional graphite/polysulfone (Gr/Ps) tape. This layup produced a nearly "isotropic" sheet with minimum total thickness. The Gr/Ps stock sheets were prefused and stored until needed. The stock sheet parts were trimmed to contour to form the skin parts (including doubler material), stacked and then postformed and fused together to produce the skin panels.

The advantages of the sheet stock approach are:

1. The sheet stock can be fabricated by vendors in large sizes on an economical basis with automated equipment.

2. The sheet stock has no storage life restrictions and can be inventoried in the same manner as metallic sheet stock, available for immediate usage after a brief oven-drying cycle.

3. The sheet stock can be readily inspected for resin content, fiber orientation, and properties. Once inspected, resin content and fiber orientation remain fixed.

4. Laminate consolidation is accomplished in making the sheet stock. Therefore, thick laminates can be made and formed with relatively little movement of the fibers.

5. Rigid sheet stock can be easily and rapidly handled on a production basis. Tacky prepreg, especially in larger sheets, require two or more persons for layup. In most cases, sheet stock can be handled and positioned by one person.

The sheet stock was made up from 3-in. wide unidirectional tapes as shown in Figure 5-1. During layup a heating iron was used to tack the material in place. When completed the stock sheet was bagged between two sheets of .030-in. stainless steel. The material was then fused and consolidated in an autoclave at 600°F and 200 psi, resulting in a sheet of flexible flat material similar to that shown in Figure 5-2. The sheet stock was then stored until needed.

Standard ultrasonic inspection techniques can be used for stock sheet quality assurance. Figure 5-3 shows a defective piece cut from a Gr/Ps stock sheet and its signature from an ultrasonic through C-Scan (obtained in a related Boeing IRAD program referred to in Section 8.3). The defect, a debonded outer ply, is clearly discernible in the scan and is the result of

Figure 5-1. Gr/Ps Sheet Stocks Layings

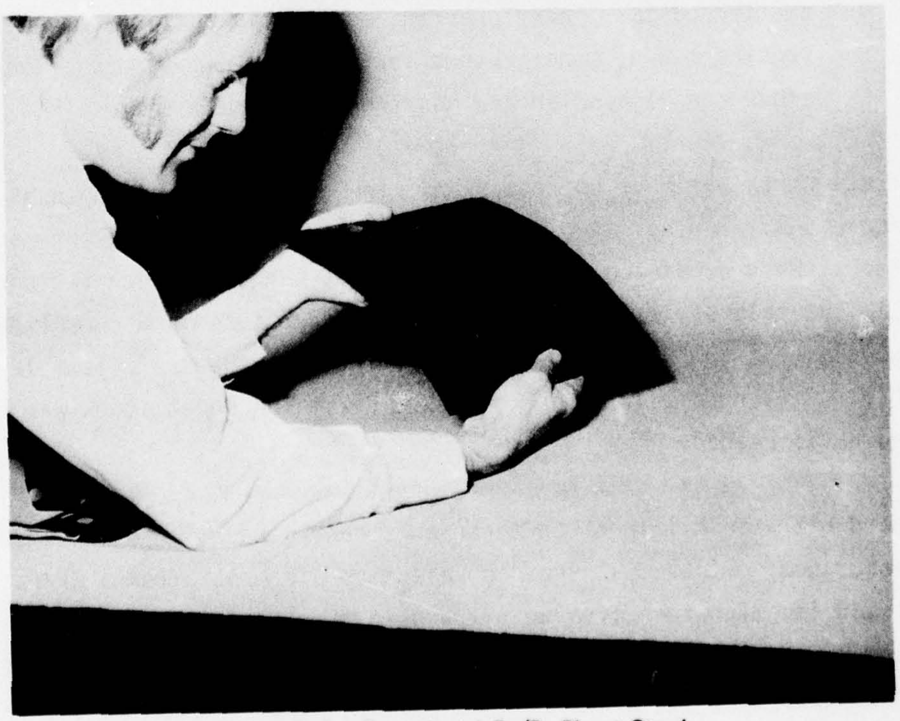

Figure 5-2. Completed Gr/Ps Sheet Stock

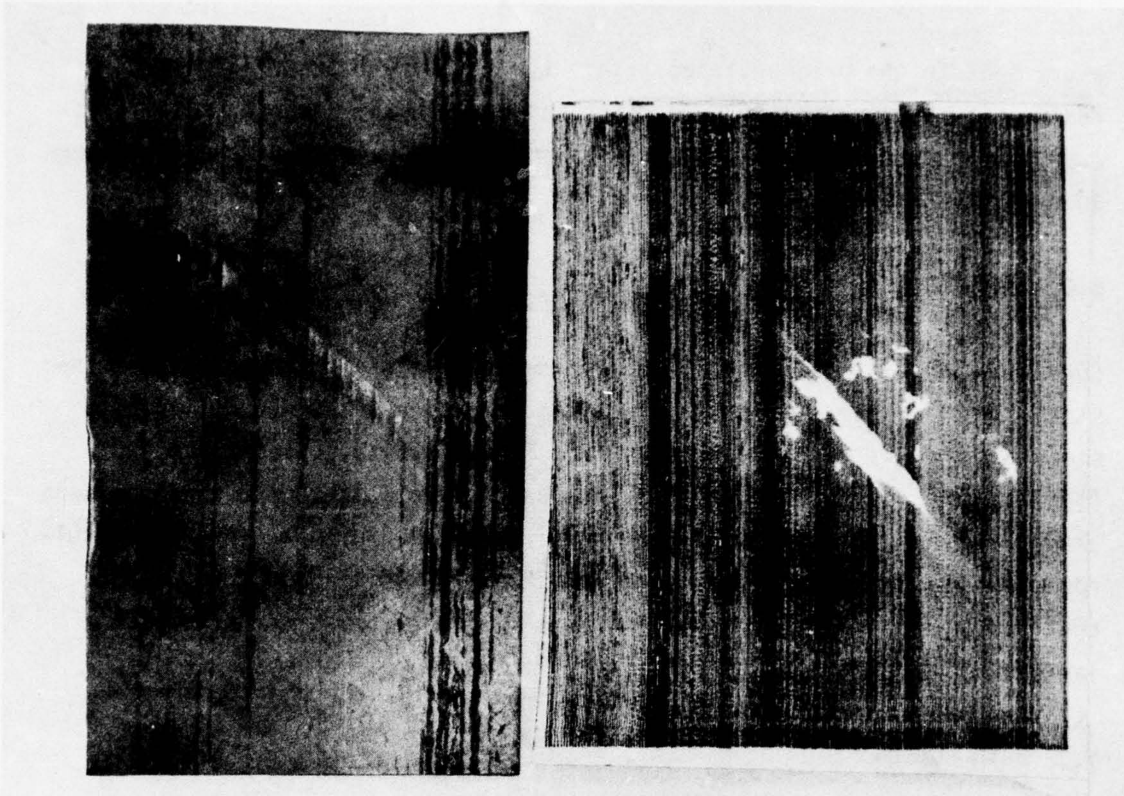

Figure 5-3. Gr/Ps Stock Sheet Defect (Outer Ply Debonded) and Ultransonic Scan

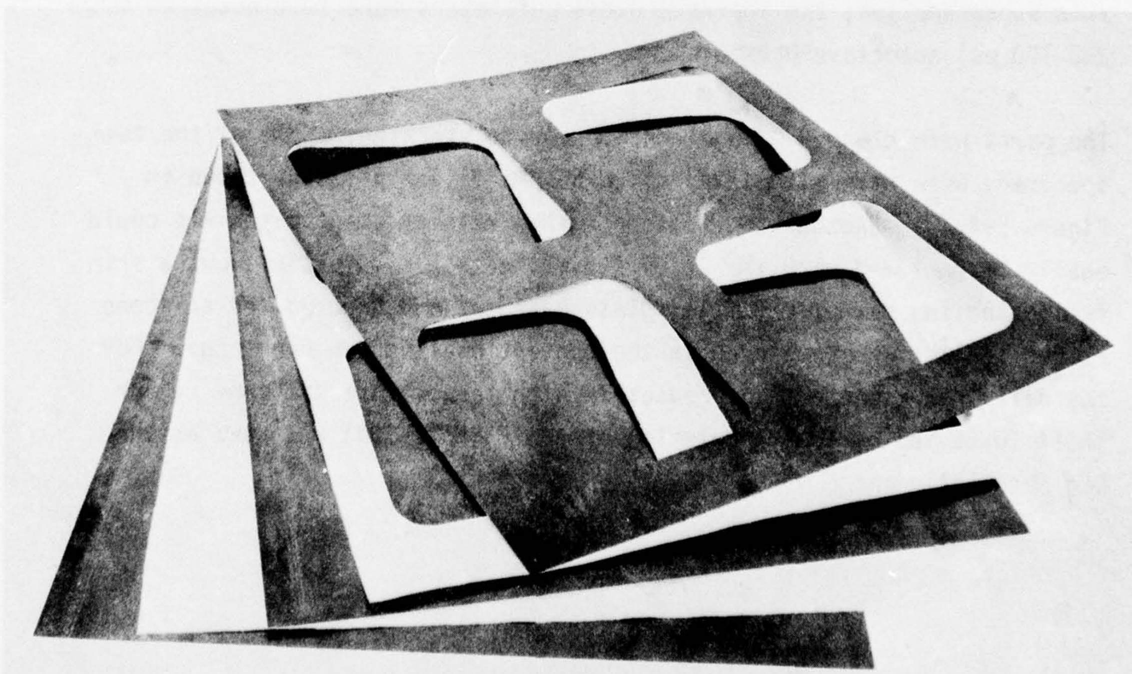

Figure 5-4. Gr/Ps Stock Sheets and Interleaved S-Glass Fabric/Ps Plies

a dry spot in the prepreg tape. (This type of defect occurred only once in the program.) Water bath coupling was used in the scanning setup (discussed in Section 8.0) which presents no problems because of the low water absorption characteristic of Gr/Ps (Ref. 1).

5.2 PREFORMING

Curved Gr/Ps parts required for the hardware fabrication were cut from the stock sheet material and then hot formed to contour. Simple roll-formed stainless steel sheet tooling (shown in Figure 5-5) was used to control final contour. Both hand pressure and vacuum bag hold-down techniques were used to shape the parts after oven heating to 375-475°F (above the polysulfone heat deflection temperature of 345°F at 264 psi). It is important that over-heating does not occur without confining pressure; otherwise a laminate "bulking" effect will result.

5.3 SKIN FUSING

The preformed Gr/Ps skin parts were fused together to form the complete skin subassemblies; the fusing process parameters were 10 minutes at 450°F and 100 psi autoclave pressure.

The parts were cleaned with alcohol prior to stacking. Some of the test specimens were made with interleaved SGF/Ps filler plies as shown in Figure 5-4. In another application, other reinforcement materials could easily be included with the Gr/Ps stock sheets. Figure 5-5 shows a skin fusing tooling setup; FEP film, glass bleeders and uncured red silicone rubber sealant were employed in the conventional vacuum bag setup. For the delivered components, a reuseable high temperature silicone rubber sheet (0.05 in. of Boeing Material Standard BMS-1-18B) was used as the bag material.

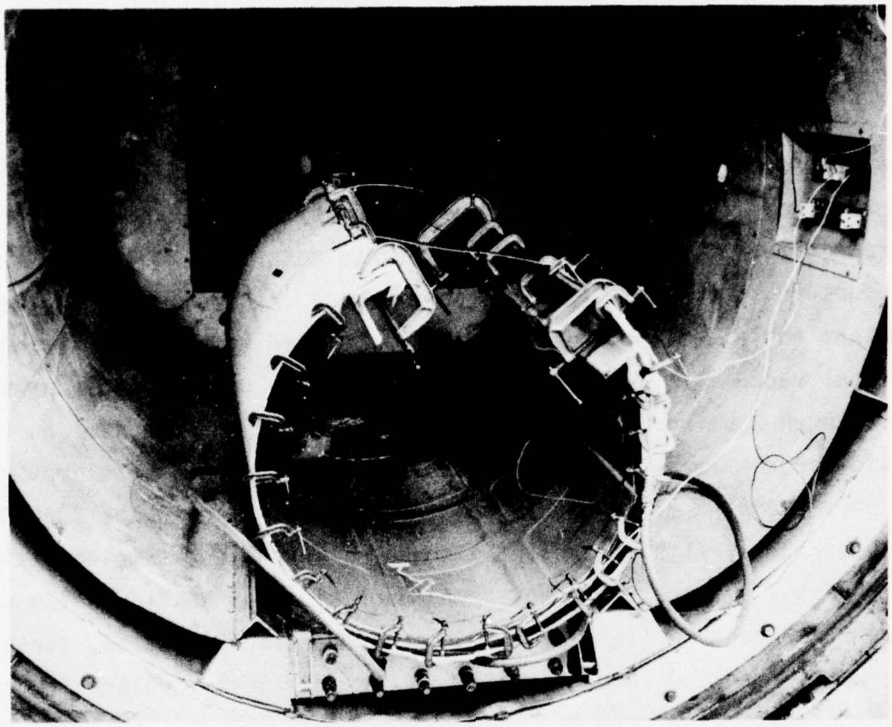

Figure 5-5. Fusing of Gr/Ps Stock Sheets Into Sub Component No. 2 Skin and Other Parts

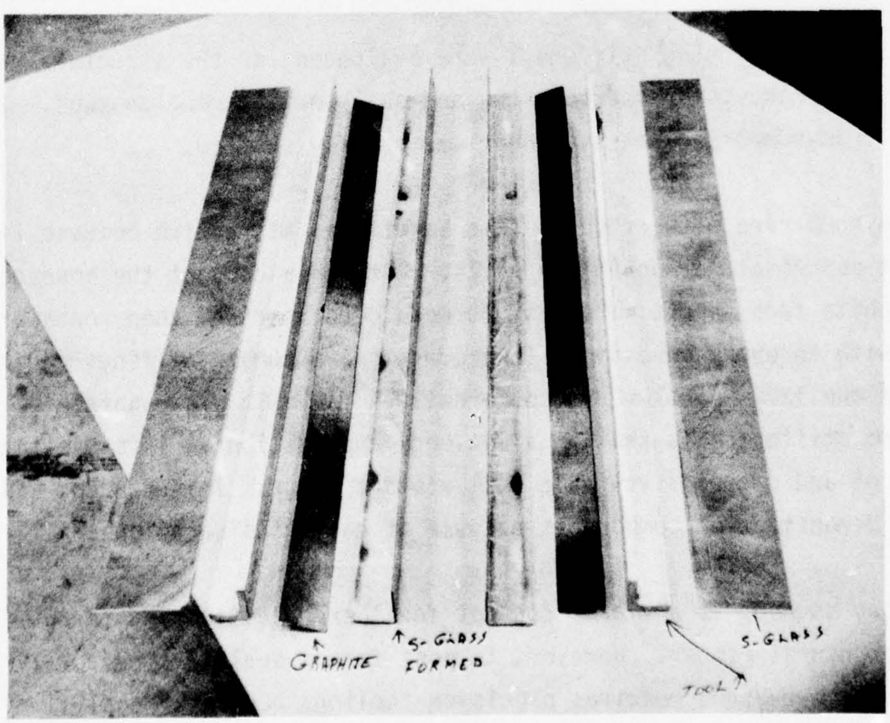

Figure 5-6. Thermoplastic Stringer Parts and Tooling Prior to Fusing

5.4 STRINGER FABRICATION

The stringer sections used in the test subcomponents and delivered components were fused from preformed SGF/Ps prepreg broadgoods and unidirectional Gr/Ps prepreg. In a production program, these materials would be purchased from vendors in stock sheet form and postformed. Figure 5-6 shows the simple aluminum angle tooling and composite parts used in stringer fusing. conventional vacuum bag setups (with FEP film) were used to apply pressure to the stringer tooling. The autoclave fusing cycle was the same as the skin fusing process. Alignment bolts at the tooling angle ends provide initial part clamping. The tooling angles were machined to the XBQM-34E inside skin radius, with a bondline allowance, for a good skin-to-stringer fit. Figure 5-7 shows a representative fused stringer section which was used on subcomponent No. 1. In a low-cost production situation, stringers would be postformed and fused in metal matched die press tooling instead of the angle and bag setup used for the prototype parts.

5.5 RING STIFFENER FABRICATION

Two basic types of ring stiffeners were evaluated for the structural rings. These were a honeycomb sandwich ring and an I-section ring concept. In addition, a foam-supported hat stiffener was briefly studied.

The honeycomb ring with graphite face sheets was attractive because it showed low-cost potential for prototype parts. With this concept the honeycomb core and graphite face sheets were post formed to contour and then secondarily bonded with an epoxy adhesive. In production, a number of rings could be cut from one large panel very economically. The main disadvantage of the honeycomb stiffeners is that in a wet-body application it is subject to fuel entrapment and occupies valuable fuel storage space. This concept was chosen for the demonstration components because of cost considerations.

The I-ring concept is a viable concept for thermoplastics and is an efficient structural memeer. However, to post form a quality structural ring to a double curvature requires precision tooling. Concept feasibility was

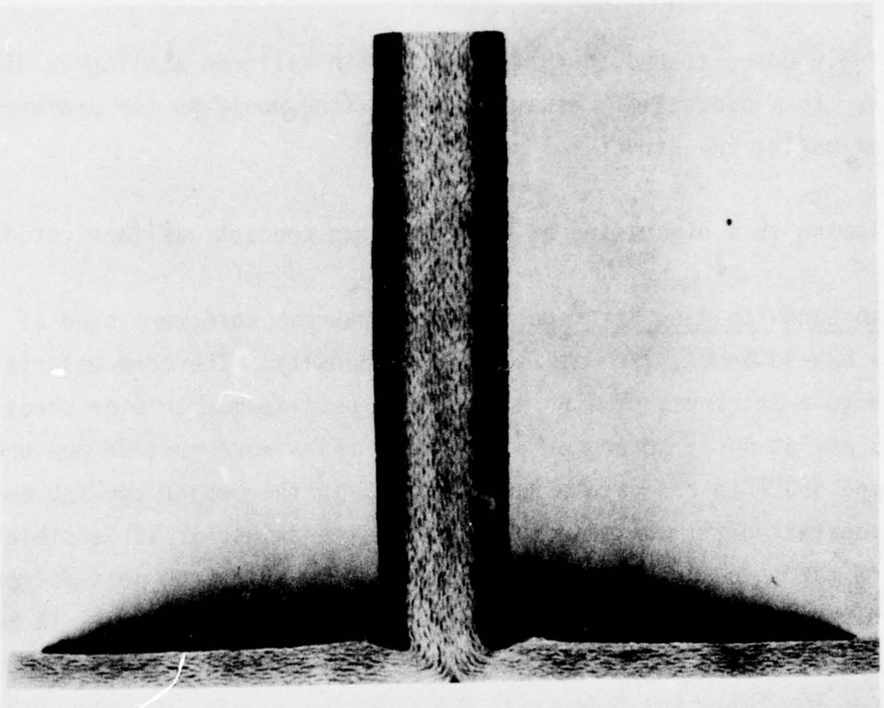

Figure 5-7. Fused Stringer Cross-Section

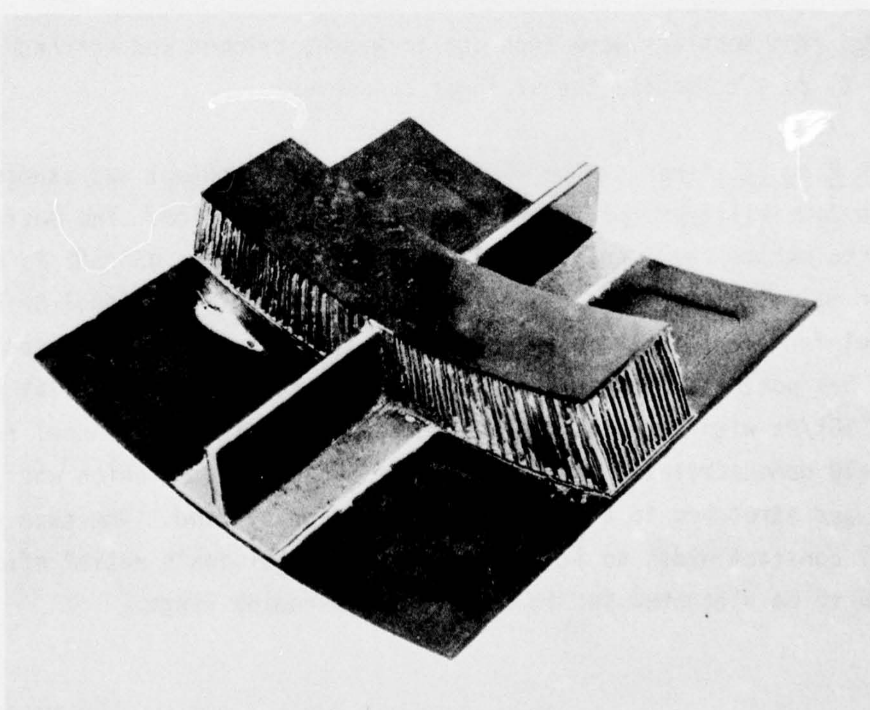

Figure 5-8. Honeycomb Sandwich Ring Cross-Over at Stringer

successfully demonstrated in this program with silicone tooling as discussed below. In a production situation, the I-ring would be the preferred method of making the structural rings.

The following is a discussion of how each ring concept was fabricated.

Honeycomb Sandwich Ring Stiffener -- The honeycomb core consisted of Hexcel's HRH-10 Nomex, 1/4-in. cell, 4-pcf density. The core material was machined to a thickness of 1.5 in. and then post-formed in wide sheet form under 15 psi at 500°F to contour. The face skins were post-formed under 50 psi and 450°F in a separate operation. With the proper tooling and a Kapton separating film between the core and the skins, it is feasible to post-form the core and skins in one operation. Having the post-formed core and skins to contour, an epoxy adhesive film (3 M's AF-143, 14 mil) was placed between the core and skins and the assembly secondarily bonded together. The following cure cycle was used:

1 hr @ 325°F, 30 psi (5° per min rise and cool down)

The bonded ring sections were then cut to width, trimmed and notched (see Figure 5-8) to accommodate the stringer crossovers.

I-Section Ring Stiffener -- The I-section stiffener concept was studied to establish feasibility for a future manufacturing situation. The basic approach to making the I-section was to first post-form $\pm$ 45° SGF/Ps prepreg over male tooling, into channel sections with unidirectional Gr/Ps flange reinforcement included as shown in Figure 5-9. The sheet stock approach has potential for curved ring stiffeners; for example, flat sheet stock of SGF/Ps with 45° bias could be cut and hot-sized to channel form. Figure 5-10 demonstrates the formability of SGF/Ps prepreg which was heated to 500°F and stretched to a 30 percent elongation by hand. The tape was initially constant width so it is apparent that "Poisson's ratio" effects will need to be accounted for in forming tight-radius rings.

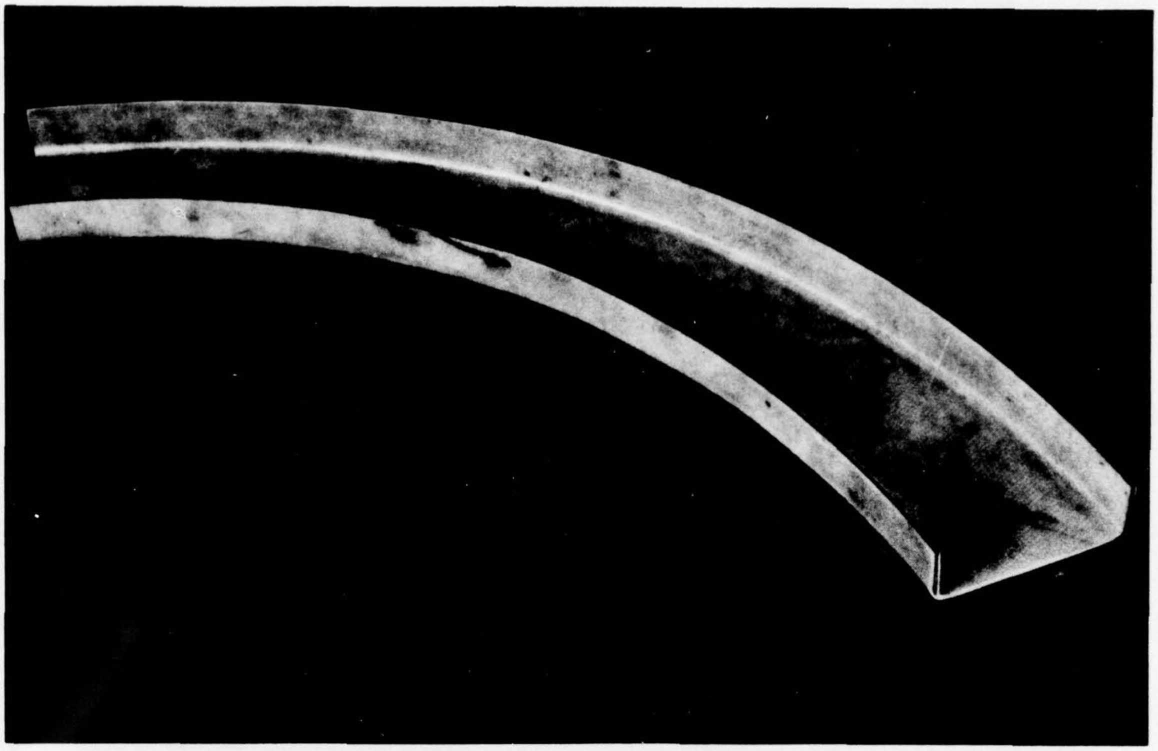

Figure 5-9. Gr/Ps and S-Glass Fabric/Ps Channel Section

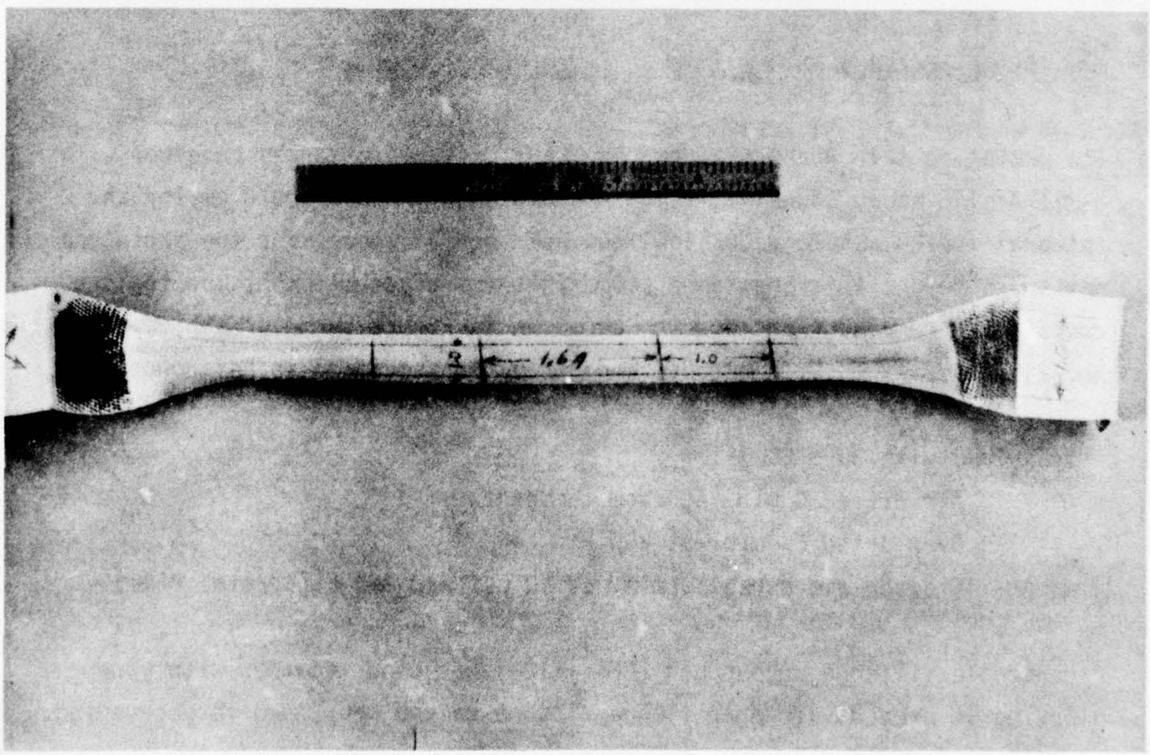

Figure 5-10. Hot Stretched S-Glass Fabric/Ps Pre-Preg Strip

In a second fusing operation, two channels sections would be joined to form the final I-section. In the interests of low-cost, a short I-section was autoclave-fused in cast silicone rubber tooling as shown in Figure 5-11. The silicone rubber used was Dow 630A with RTV-630B curing agent.

Another stiffener concept that has possible use in rings and stringers was briefly investigated -- a foam supported hat stiffener. Figure 5-12 shows two hat stiffeners made from the Gr/Ps stock sheet discussed previously. Both sections were post-formed to shape around blocks of 12 percent 1100 aluminum foam (Duocel produced by Energy Research and Generation, Inc., Oakland, California) and simultaneously fused to a Gr/Ps stock sheet simulating a skin section. One of the sections in Figure 5-12 has the foam removed by chem-milling leaving an open section. A specific application would determine whether the foam would remain or be removed. The foam material is of interest because it can survive the fusing temperatures and pressures associated with polysulfone. The foam stiffener concept was identified late in the program and thus was not incorporated in the component hardware.

5.6 FINAL ASSEMBLY

The prototype skin and stiffener component parts were bonded together with 7-mil AF-143 epoxy adhesive film. A production process would employ an integral fusing assembly for low-cost but such a process for the prototype parts was beyond this program's scope in terms of development and tooling costs. The component parts, such as shown in Figure 5-13 for subcomponent No. 2, were cleaned with alcohol and bonded as specified in Reference 4.

Primed with 3M's EC-3917
    Air dried 30 min. at room temperature
    Oven dried 60 min. at 250°F
AF-143 autoclave cured 60 min. at 50 psi and 350°F (5°/min. rise).

A reuseable silicone rubber bag (BMS 1-18B) was used together with wood blocking to prevent stringer flange distortion and honeycomb core crushing.

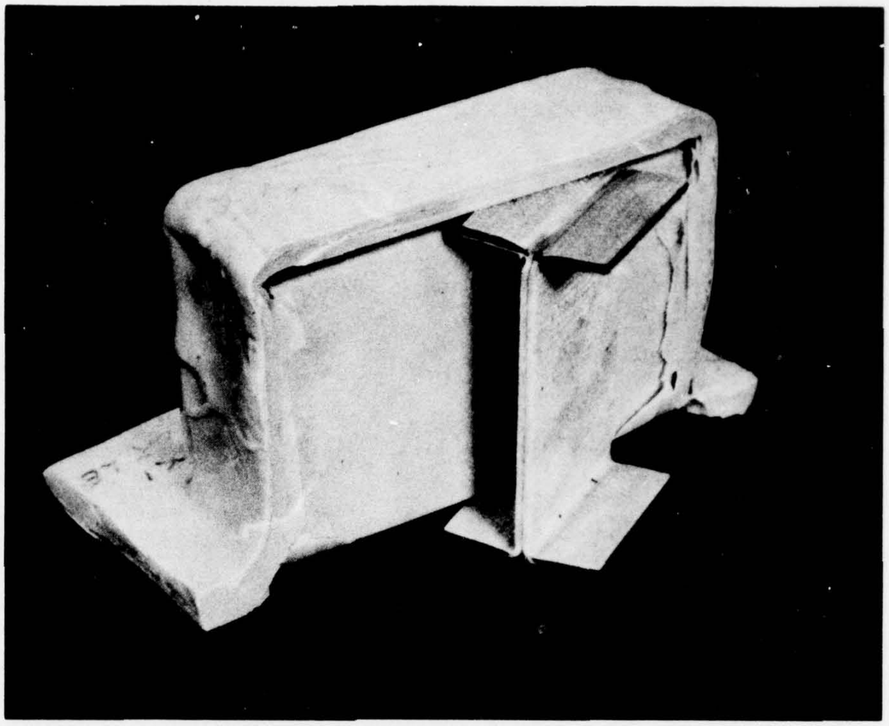

Figure 5-11. I-Section Fused From Gr/Ps and S-Glass Fabric/Ps Channels

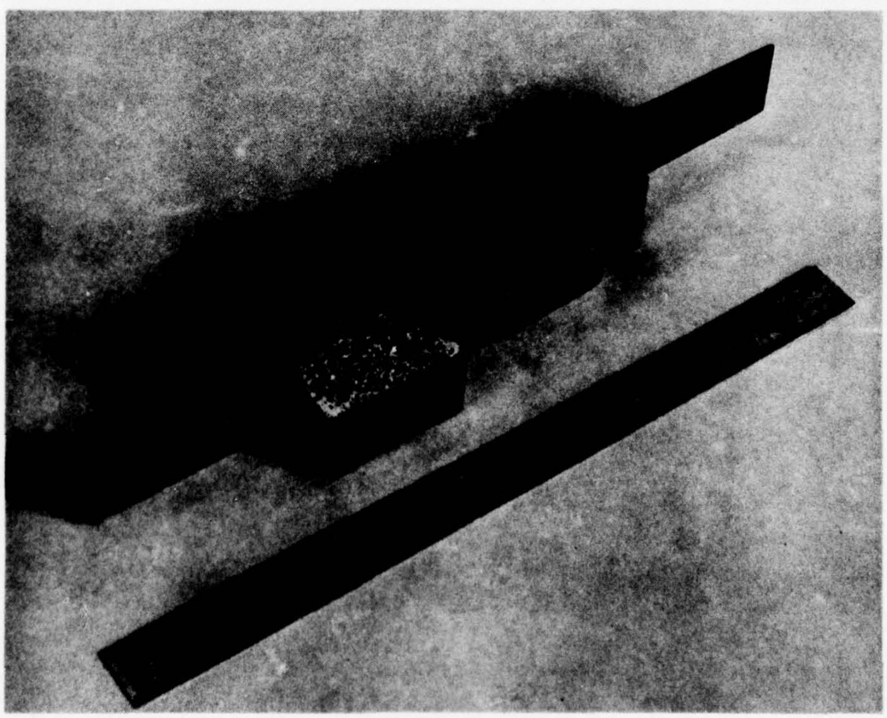

Figure 5-12. Aluminum Foam – Gr/Ps Thermoplastic Hot Stiffener Concept

A conventional epoxy bonding vacuum bag setup was used. Figure 5-14 displays a bonded assembly after cleanup.

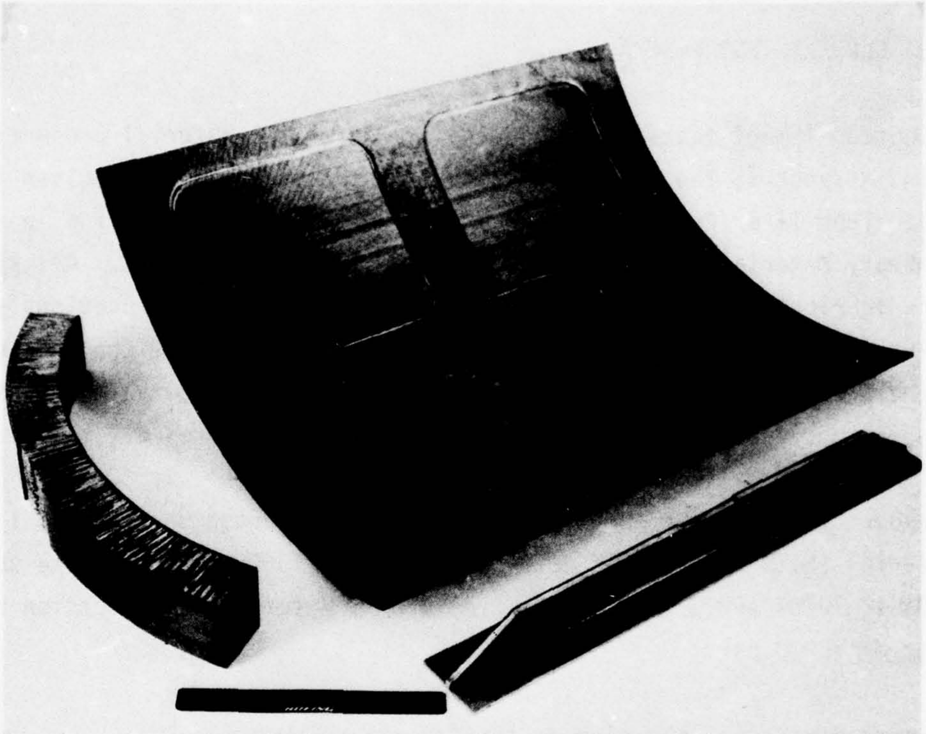

Figure 5-13. Parts for Sub Component No. 2

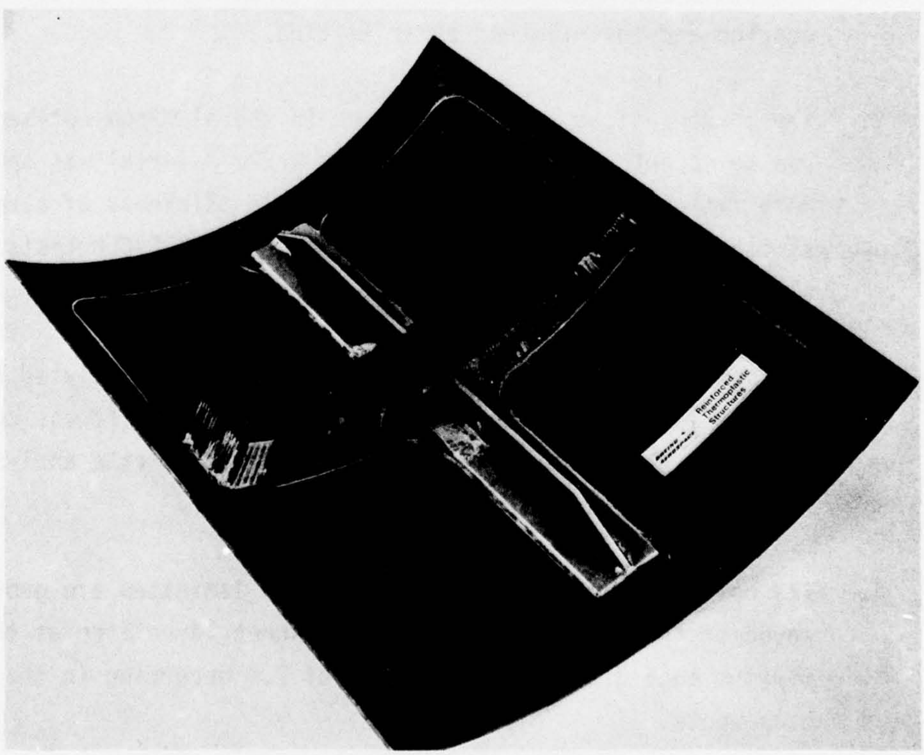

Figure 5-14. Bonded Sub Component No. 2 Assembly

6.0 ELEMENT TEST PROGRAM

Selected element tests were conducted to determine material properties and joint strengths; the results of the element test program are given in Figure 6-1. Type II Gr/Ps material (Hercules' 3004/A-S), as specified in the Preliminary Material Specification, was used for the specimens. All specimens were fabricated by the fused stock sheet method described previously. The joint specimens were three stock sheets thick to simulate the component joint areas. Other specimens were one or two stock sheets thick, as noted. The various specimen configurations are shown in Figures 6-2 to 6-5.

Standard interlaminar shear tests were also conducted on the Type I GR/Ps material (U.S. Polimeric's 3023) and averaged 10026 psi for three specimens (10301, 10007 and 9720) vs. the Preliminary Material Specification requirement of 10000 psi.

Several aspects of the element testing are noteworthy:

1. The Gr/Ps material exhibited nonexplosive failure in the bolt bearing and interlaminar shear testing.

2. The joint fatigue specimens failed in the aluminum net-sections due to eccentric load effects. The Gr/Ps material was apparently more fatigue resistant than a comparable thickness of aluminum (a similar finding occurred in high cycle NAVAIRDEVCEN tests on aluminum-to-Gr/Ep bolted joints, Reference 5).

3. The bending stiffnesses (shown in Figure 6-1) calculated from the flexure test results compare closely with the stiffness computed for equal thickness laminates by classical laminate analysis (these stiffnesses are given in Figure 9-5).

4. The bolt bearing strengths of the Gr/Ps laminates are good because of the nearly isotropic stock sheet layup even at the minimum edge distance ratio (e/d) of 2.0 occurring in the Gr/Ps components.

Category	Specimen No.	Temp (^oF)	Failure load (lb)	Cycle No.	Notes
1. Joint tension in 0^o direction	JT-0-1	RT	5,060	1	• 3 Gr/Ps stock sheets @ 0^o + 2 SGF/Ps/181 style interleaved plies @ 45^o • Width = 1.5 in. • 4-0.1875 in CSK huck bolts at each end fastening to 0.1875 7075-T6 aluminum end tabs
	-2	RT	5,020	1	
	-3	200	5,270	1	
	-4	200	5,570	1	
2. Joint tension in 90^o direction	JT-90-7	RT	4,450	1	• Same as above except for laminate orientation
	-8	RT	4,655	1	
	-10	200	4,605	1	
	-11	200	4,770	1	
3. Joint tension fatigue in 0^o direction	JT-0-5	RT	1,500	285000	• Same configuration as above
	-6	RT	1,500	492000	• R = 0.1 cyclic loading
4. Joint tension fatigue in 90^o direction	JT-90-9	RT	3,000	55000	• Same configuration as above
	-12	RT	3,000	62000	• R = 0.1 cyclic loading

Category	Specimen No.	Temp (^oF)	Failure stress (lb/in.^{2})	Elastic modulus E (lb/in.^{2})	Bending stiffness D (in.-lb)	
5. Tension	T-0-1	RT	57,700	7.5E6	—	• Gr/Ps stock sheet @ 0^o
	-2		75,600	7.1E6		• Width = 1.0 in.
	-3		71,000	7.8E6		• Bonded fiberglass end tabs
	-4		68,600	6.5E6		• No failures in grip area
	-5		67,600	6.0E6		
	-6		67,800	7.3E6		
6. Flexure in 0^o direction	F-0-1	RT	162,800	—	232.1	• 2 Gr/Ps stock sheets @ 0^o
	-2		101,800		229.1	• Width = 1.5 in.
	-3		102,100		220.8	• Span = 4.0 in. • Center loaded
7. Flexure in 90^o direction	F-90-4	RT	91,200		135.8	• 2 Gr/Ps stock sheets @ 90^o
	-5		89,700		122.0	• Width = 1.5 in.
	-6		94,000		120.2	• Span = 3.0 in. • Center loaded
8. Bolt bearing in 0, 45 and 90^o directions	BB-0-1	RT	70,625	e/d = 2.0	—	• Same laminate as joint tests
	-2		60,000	4.9		• 0.1875 bolt shank in double shear
	BB-45-1		66,250	5.0		• Hand tightened nut
	-2		55,625	5.1		• Width = 1.5 in.
	BB-90-1		58,125	2.0		
	-2		58,125	5.0		

Figure 6-1. Element Test Results

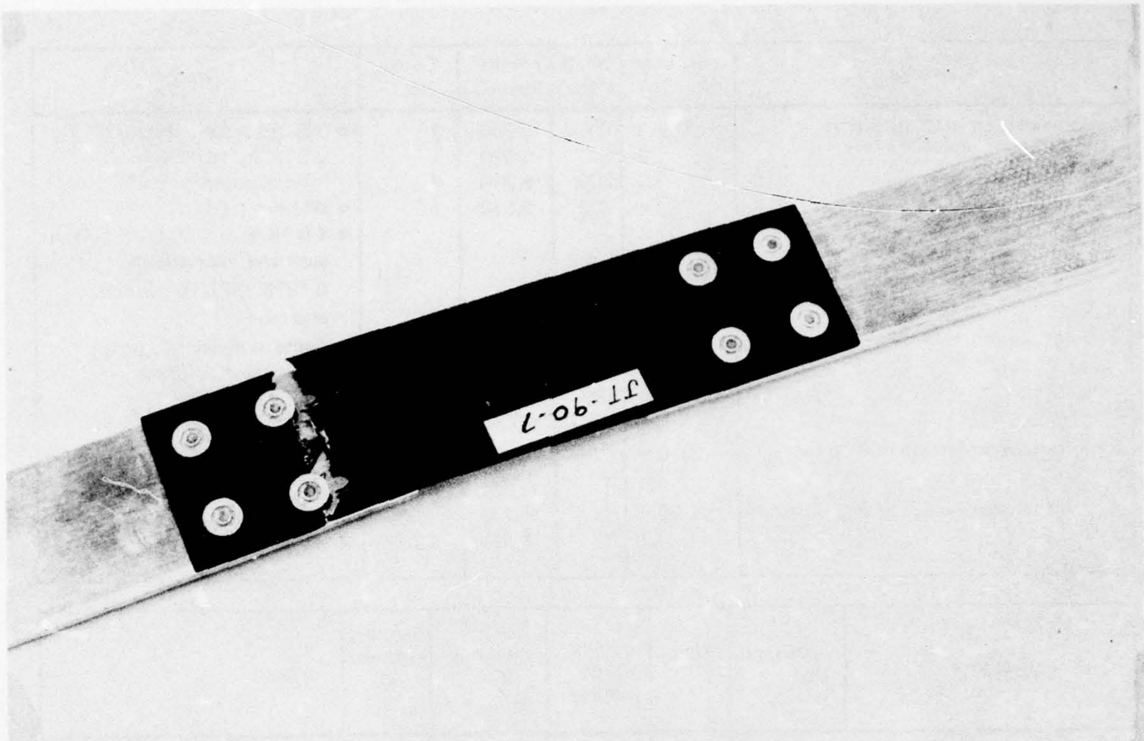

Figure 6-2. Joint Tension Specimen

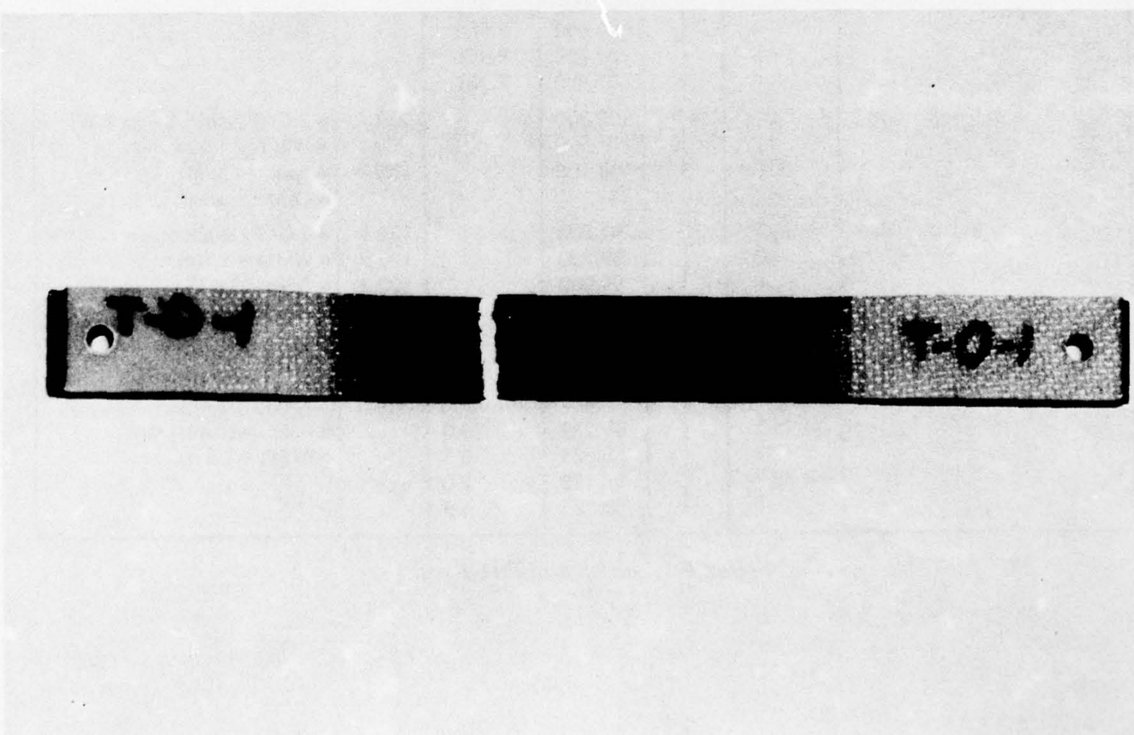

Figure 6-3. Tension Specimen

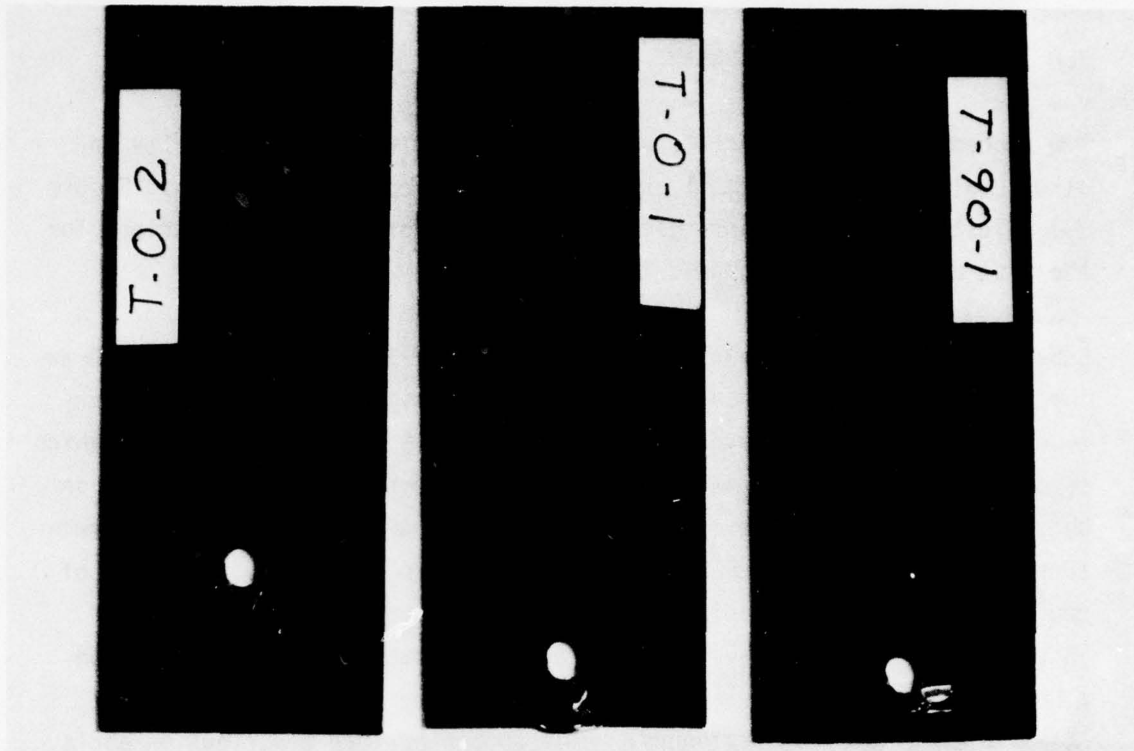

Figure 6-4. Bolt Bearing Specimens

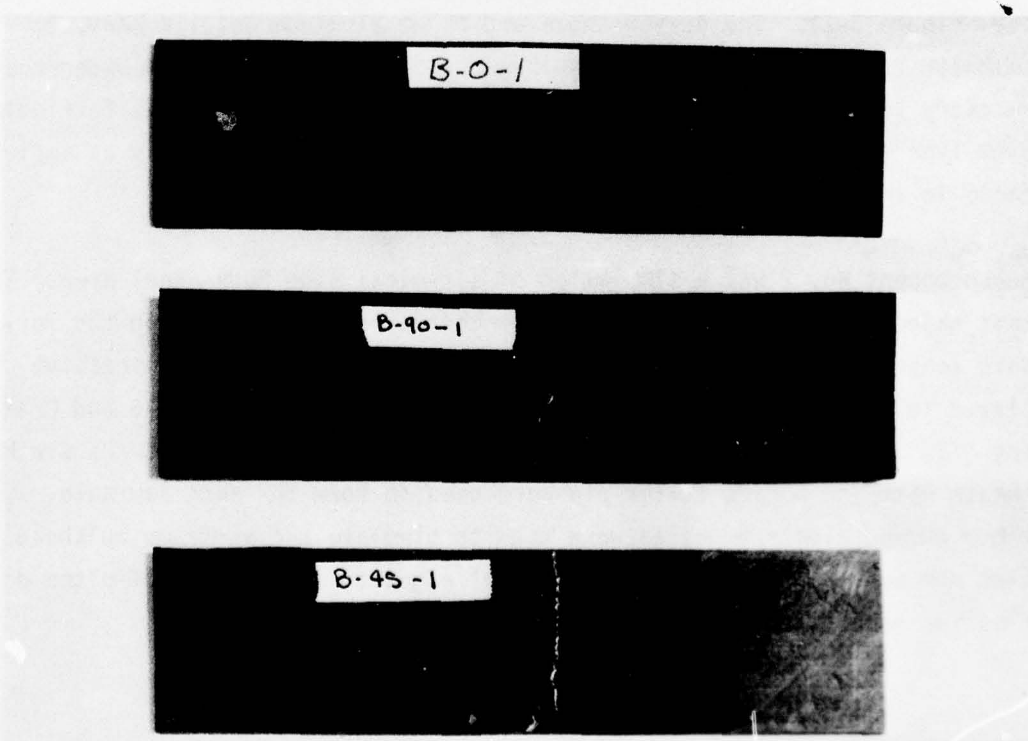

Figure 6-5. Flexure Test Specimens

7.0 SUBCOMPONENT TEST PROGRAM

Two subcomponent tests were conducted to confirm fabrication quality and structural adequacy in detail areas of the side component designs. Figure 7-1 illustrates the type of tests that were performed; detail drawings for the test articles are included in Appendix A.

Subcomponent No. 1 is shown in Figures 7-2 and 7-3 and is detailed in Drawing SK21775KO. The purpose of this test was to verify the load introduction details in the forward centerbody bulkhead area (sta. 233.50). A high concentrated load exists in this area from the forward side body longeron because of the transition from the forward half circle body plus nonstructural air inlet duct to the full circular centerbody section. Because of concerns of eccentric loading and local strain concentrations, the load introduction area was heavily reinforced with doubler sheet stock and an aluminum angle load distribution detail under the load point (the angle detail with single row fasteners). The companion load distribution angle simulated an angle longeron that is retained on the stripped-down airframe (see Figure 3-2). The curved angle and sheet aluminum details serve to simulate the airframe's forward bulkhead and ring frame located under the recovery lug at sta. 241.78. The skin of subcomponent No. 1 was fabricated from Type I Gr/Ps sheet stock material and one SGF/Ps filler ply as indicated in the detail drawing.

Subcomponent No. 2 was a simulation of a typical side body panel area. The test objectives were to (a) apply a concentrated load simulating the forward longeron load introduction and (b) apply a uniform end compressive strain to evaluate panel buckling strength. Figures 7-4 and 7-5 and Drawing SK22175KO in Appendix A define the test article. Type II Gr/Ps stock sheets with one SGF/Ps filler ply were used to form the skin laminate. Heavy curved aluminum angles were used to simulate the airframe bulkheads. Aluminum angles were bolted to the panel edges to provide out-of-plane deflection restraint.

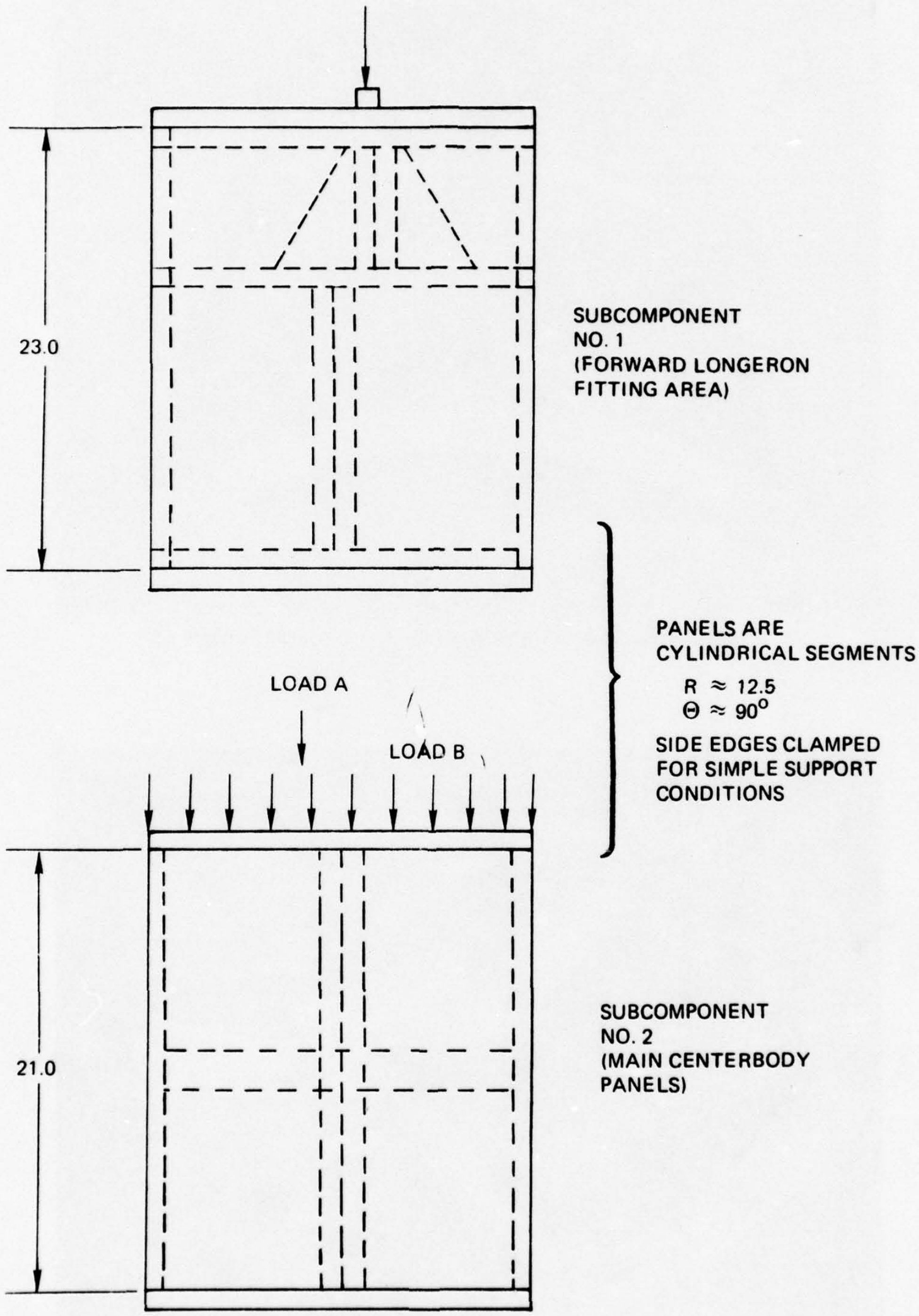

Figure 7-1. Panel Subcomponent Tests

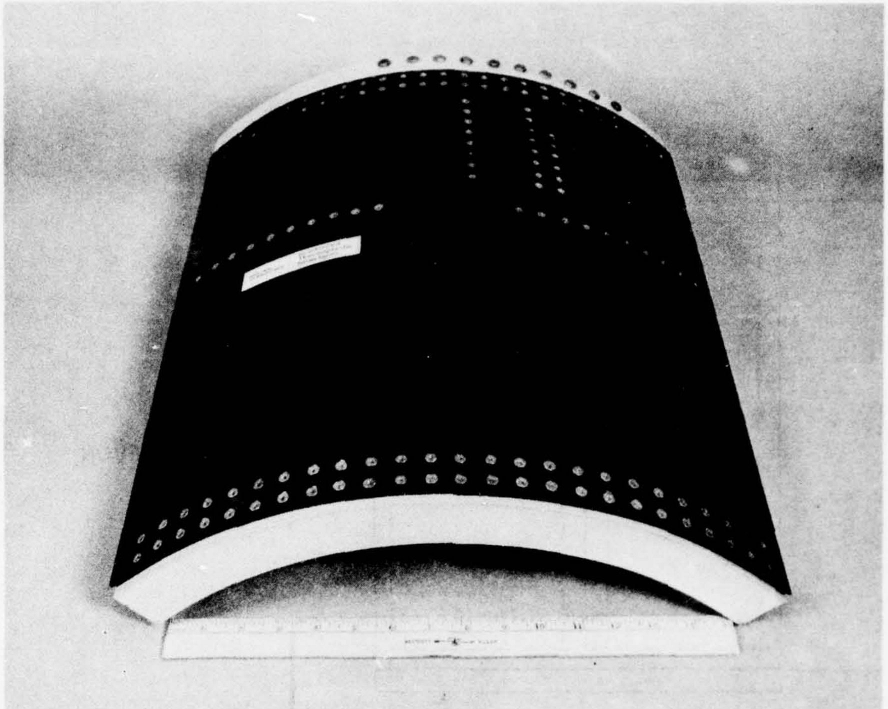

Figure 7-2. Sub Component No. 1 – Outside Surface

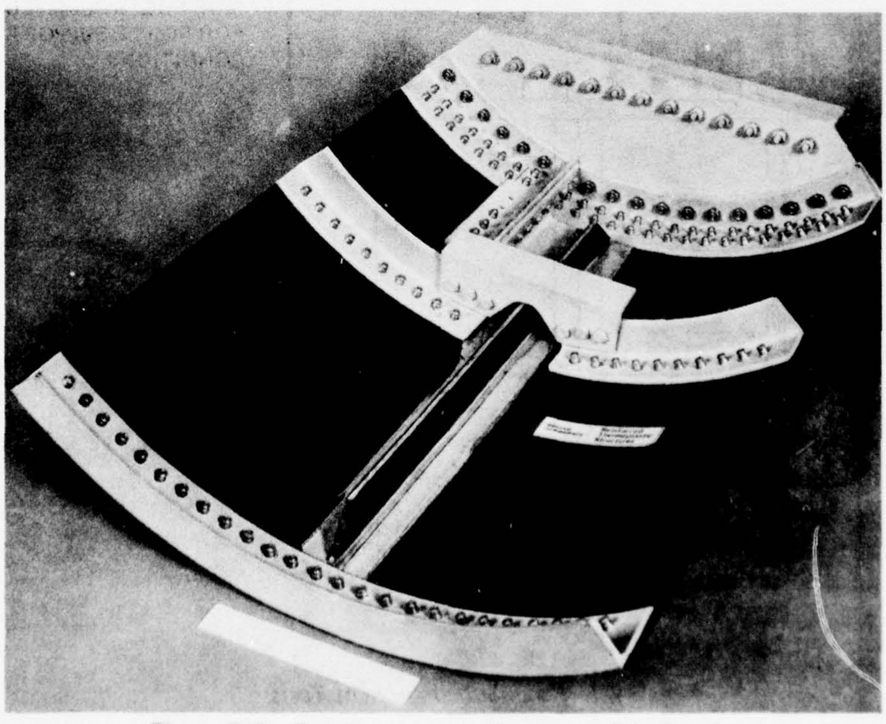

Figure 7-3. Sub Component No. 1 – Inside Details

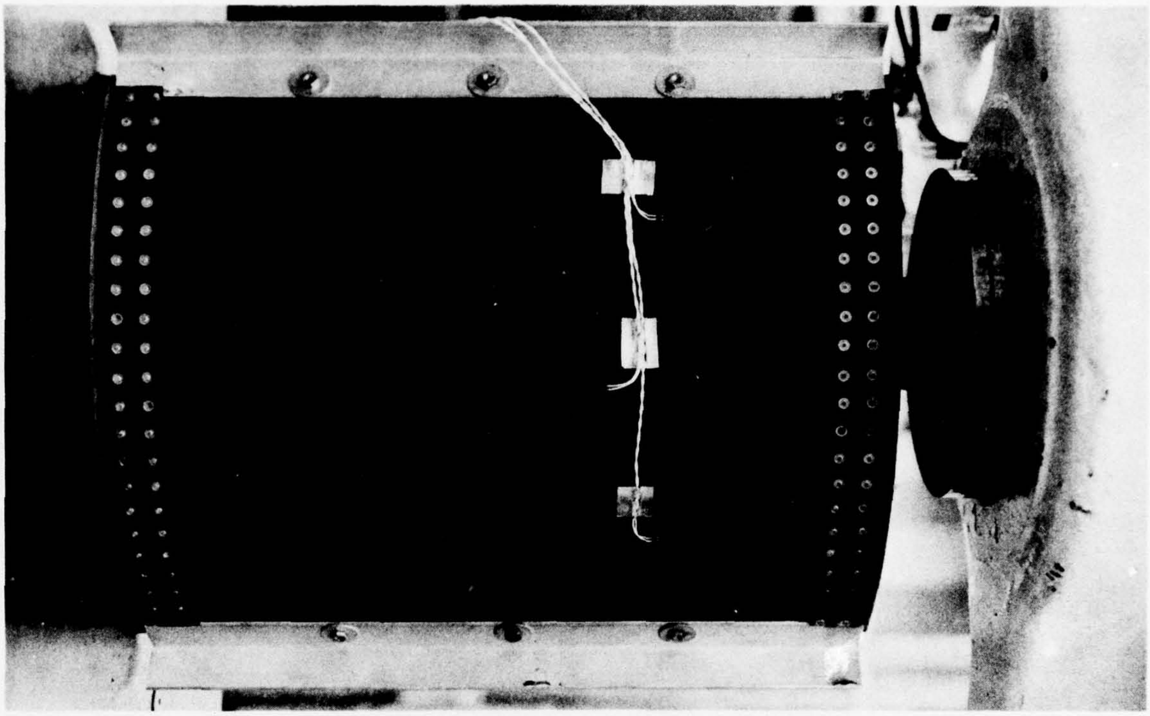

Figure 7-4. Sub Component No. 2 Outside Surface

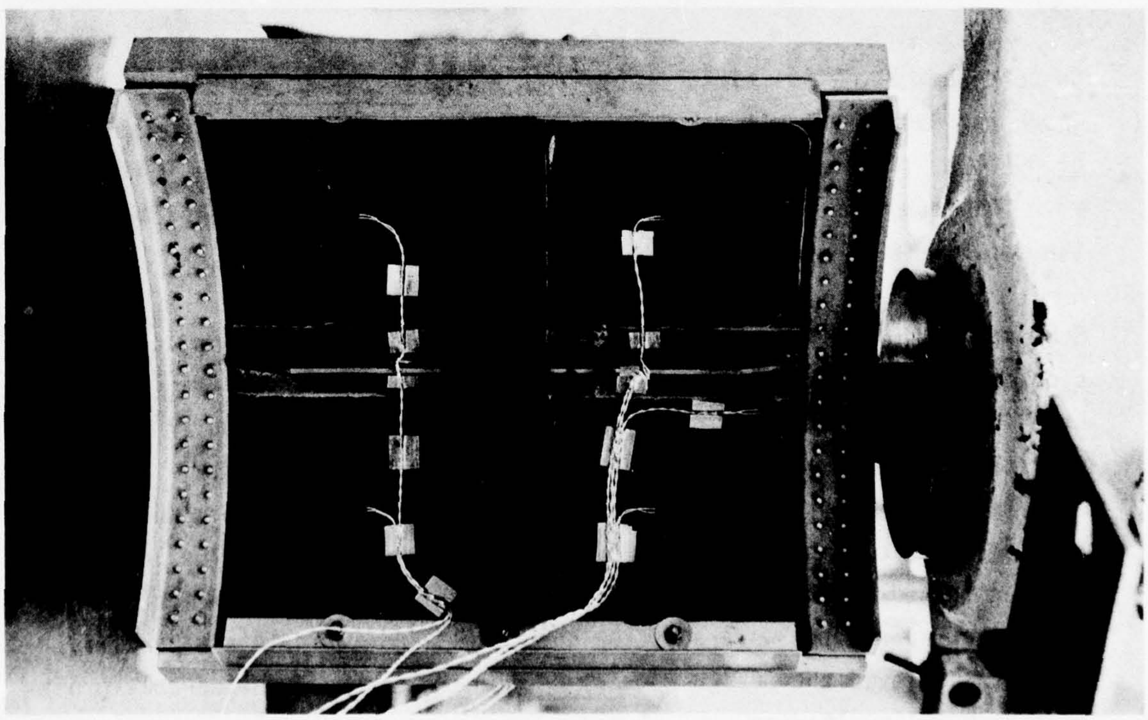

Figure 7-5. Sub Component No. 2 Inside Details

Figure 7-6 presents a summary of the subcomponent tests. The test results indicated that subcomponent No. 1 was over-strength with respect to design requirements. The article failed by prebuckling strains in the lower right panel in Figure 7-1 which caused fracturing. The load introduction area was undamaged and the measured strains were low at the failure load. The measured strains at the failure were (see Figure 7-7 for strain gage locations):

Strain gage	SG-1	-2422 microstrain ($\mu\varepsilon$, 10^{-6} in./in.)
	-2	-1931
	-3	-1861
	-4	-1527
	-5	-2238
	-6	-3660
	-7	-1880
	-8	-3609
	-9	+36
	-10	-67

Subcomponent No. 2 was first tested with a concentrated load of 6899 lb. which approached the ultimate design load of the original aluminum design of 6950 lb. (Reference 6). The load was placed off center from the stringer. The resultant measured skin strains were low and no visible change to the specimum was noted.

Subcomponent No. 2 was then subjected to a series of uniform end compression deflection loadings; the test setup is shown in Figure 7-8. An upper panel buckled elastically (snap-through) at 22551 lb. as shown in Figure 7-9 (the photograph was taken with the load applied). No visible change occurred after subsequent reloadings and buckling. The depth of buckle in Figure 7-9 is on the order of 0.5 in. (7 skin thicknesses). The loading was terminated after buckling occurred in each load cycle; however, the peak measured strains were about half of the Gr/Ps laminate's capability so it is believed the design concept could have carried considerably greater post-buckling loading.

S/C No.	Loading No.	Load lb	Max Gage No.	Strain $\mu\epsilon$	Panel Buckling Load	Notes
SC-1	11	44,682	6	-3,660	44,682	Point compression load test Panel fractured at buckling load
SC-2	2	6,899	9	-2,199	–	Point load test – no damage
SC-2	9	21,540	3	-1,964	–	Uniform end deflection (compression) Panel buckled elastically
		22,551	3	-1,875	22,551	
		21,793	3	-3,979	Post-buckling loading	
		22,551	3	-4,360		
		22,298	3	-4,416		
SC-2	11	21,625	3	-1,748	–	Repeat of above loading – no buckling
SC-2	12	21,877	3	-1,668	21,877	Repeat of above loading Panels buckled elastically
		20,363	3	-4,026	Post-buckling	

Figure 7-6. Subcomponent Test Summary

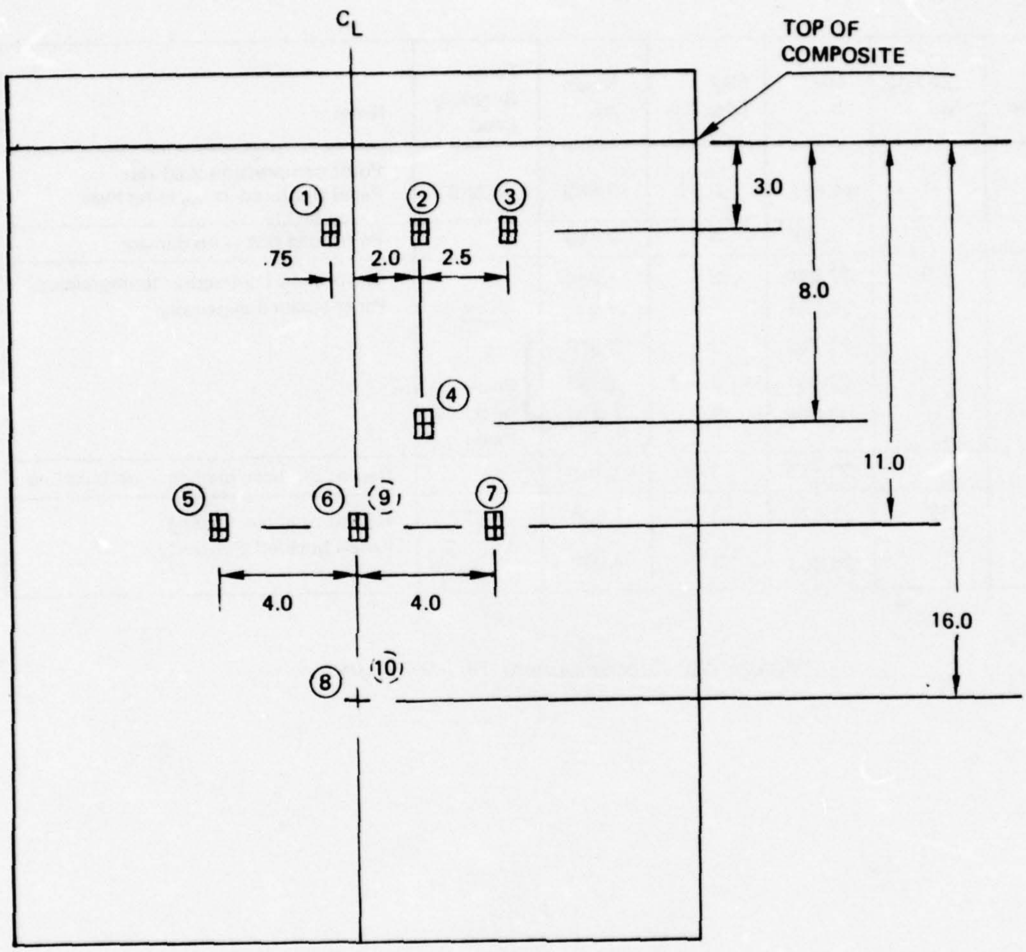

Figure 7-7. Subcomponent No. 1 Strain Gage Locations

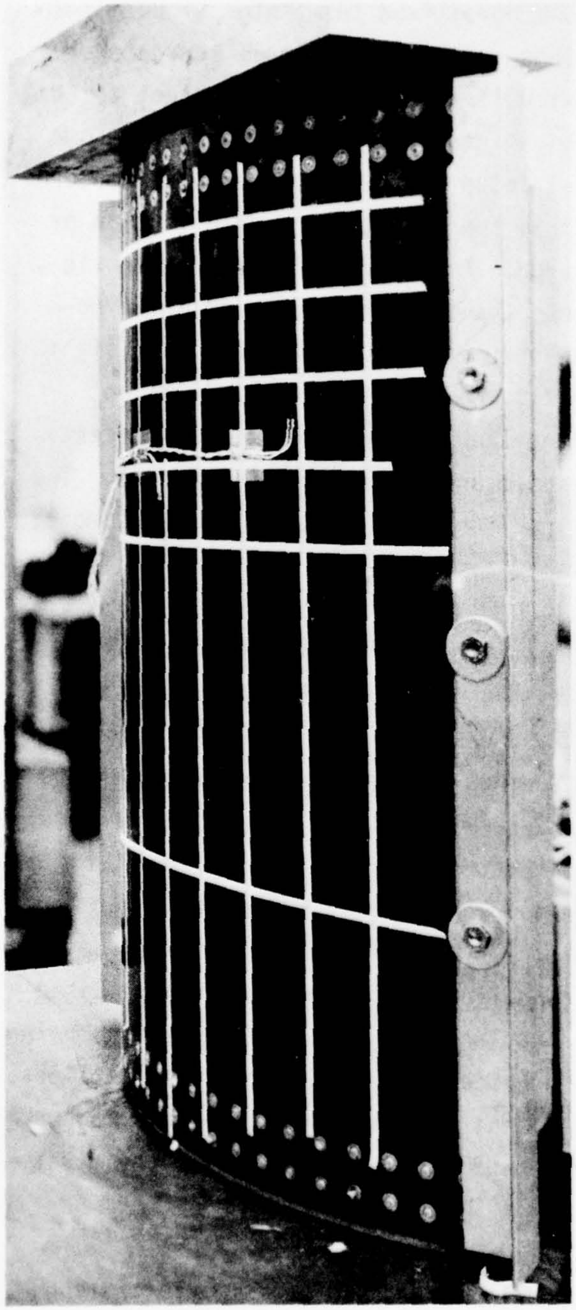

Figure 7-8. Sub Component No. 2 With No Load

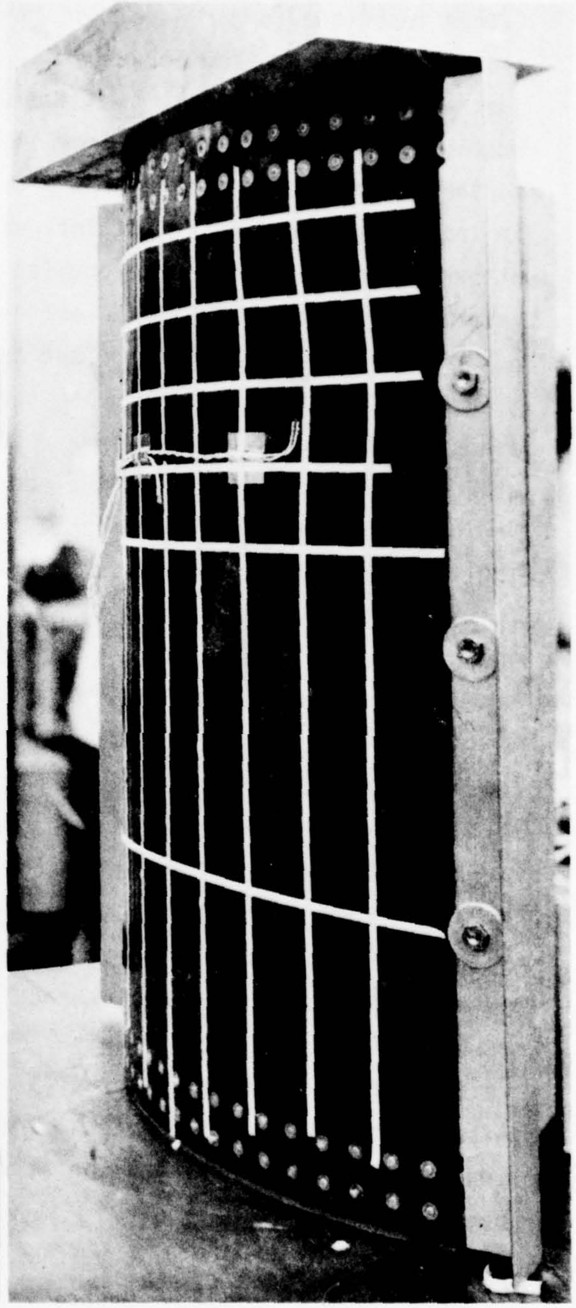

Figure 7-9. Sub Component No. 2 Under Uniform End Strain in Post Buckled Condition

After the loading tests, subcomponent No. 2 was inspected by the ultrasonic C-scan technique (under a related IRAD program) and some internal delamination was detected. The location was in a small nominal panel laminate area just adjacent to the upper side support bolt in Figure 7-9 (adjacent to the buckled panel). This was an area of high interlaminar shear which was due to the discontinuous nature of the test setup and is not an area of concern in the actual components. Of interest is the fact that no delamination or other damage occurred in the buckled area. Figure 7-10 is the C-scan signature corresponding to the defect area; the dark area represents the undamaged nominal panel laminate and the suspected damage appears as a light area sized about 1.5 x 1.0 in. on the subcomponent. Figure 7-11 is an acoustic hologram (obtained in a related IRAD program) of the same defect area in which the damage and the edge bolt and washer can be detected. The equipment used for the holographic inspection was a Holosonics brand, model 100 Thru Transmission Acoustic Holography System.

The strain gage locations on subcomponent No. 2 are shown in Figure 7-12. SG-3 and -4 were placed back-to-back in the center of the panel defect area in which the damage and the edge bolt and washer can be detected. The equipment used for the holographic inspection was a Holosonics brand, model 100 Thru Transmission Acoustic Holography System. SG-9 was located on the nominal skin laminate just under the concentrated load application area. Figure 7-13 is a plot of the SG-3 and -4 strain response during the first buckling load application; it is characteristic of a prebuckling cylindrical shell response culminating in snap-through buckling with stress reversal. Based on previous experience, the panel prebuckling response is evaluated as being relatively insensitive to out-of-plane (eccentricity, imperfection) effects. Also, the load-deflection plots of all test loadings were linear up to buckling. Figure 7-14 tabulates the strain gage data recorded during the first buckling load application.

STAGS-B computer code (Reference 7) models were coded to simulate the subcomponent test articles. These models are shown in Figures 7-15 and 7-16. The respective computer input listings define the stiffnesses calculated based on the actual gage measurements, for the various stiffeners and skins.

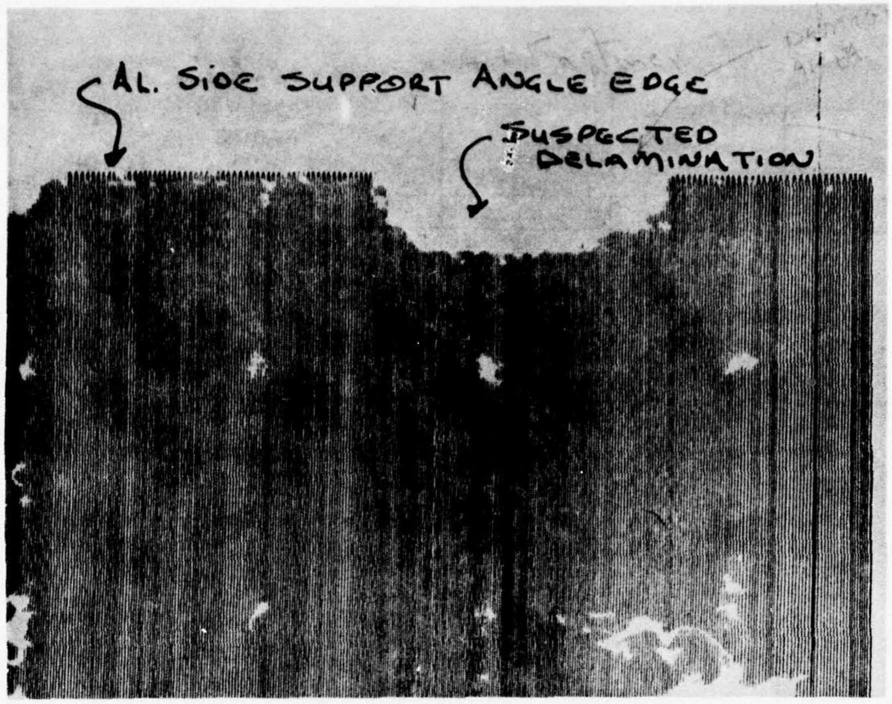

Figure 7-10. Ultransonic Scan of Side Bolt Area Near Buckled Panel of Sub Component No. 2

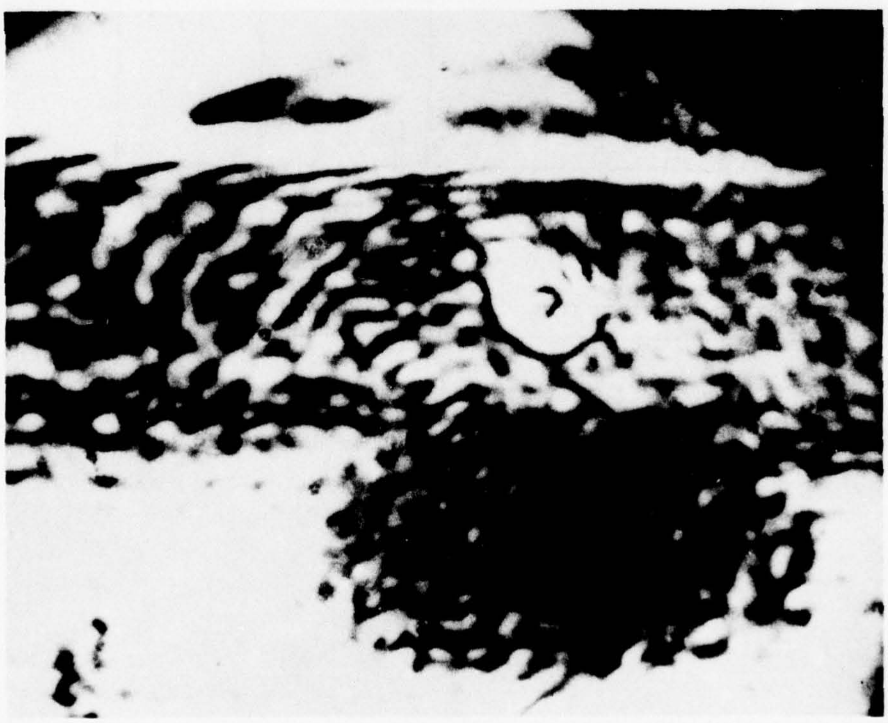

Figure 7-11. Acoustic Hologram of Side Bolt Area in Subcomponent No. 2

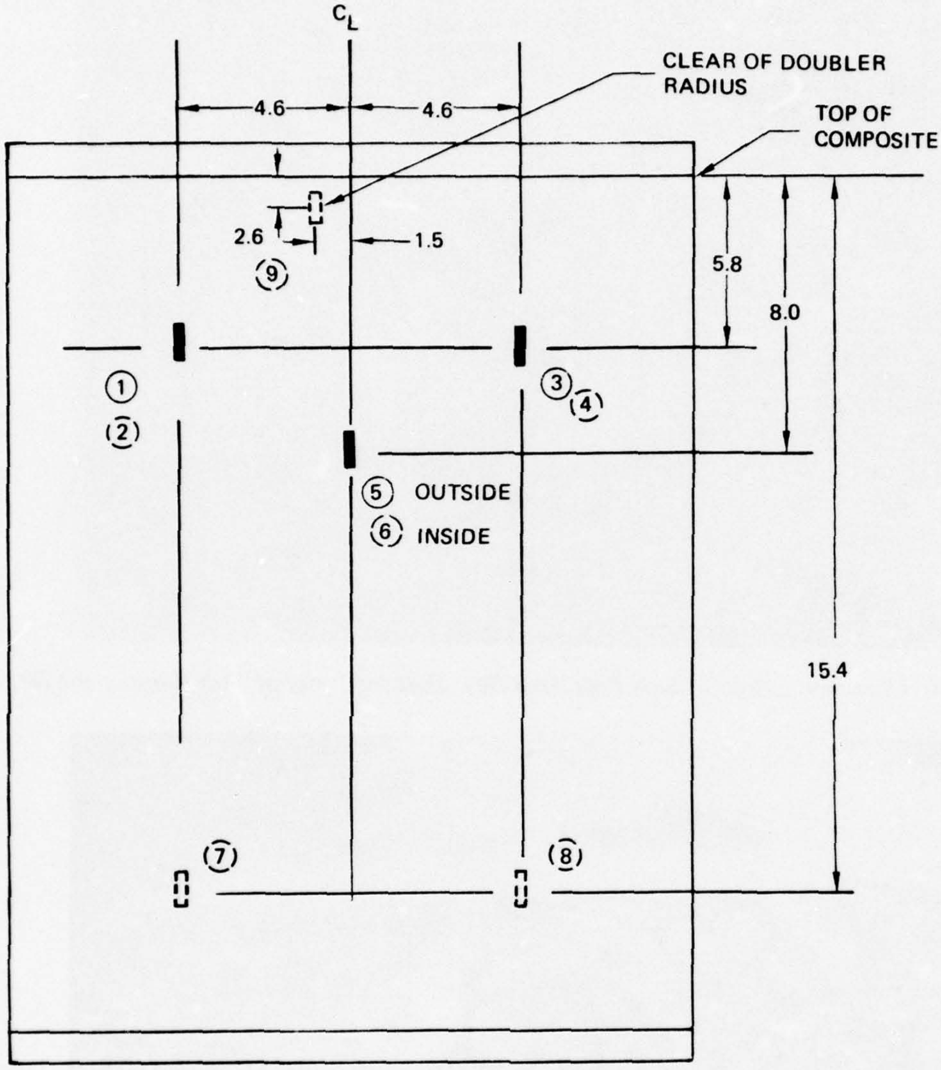

Figure 7-12. Subcomponent No. 2 Strain Gage Locations

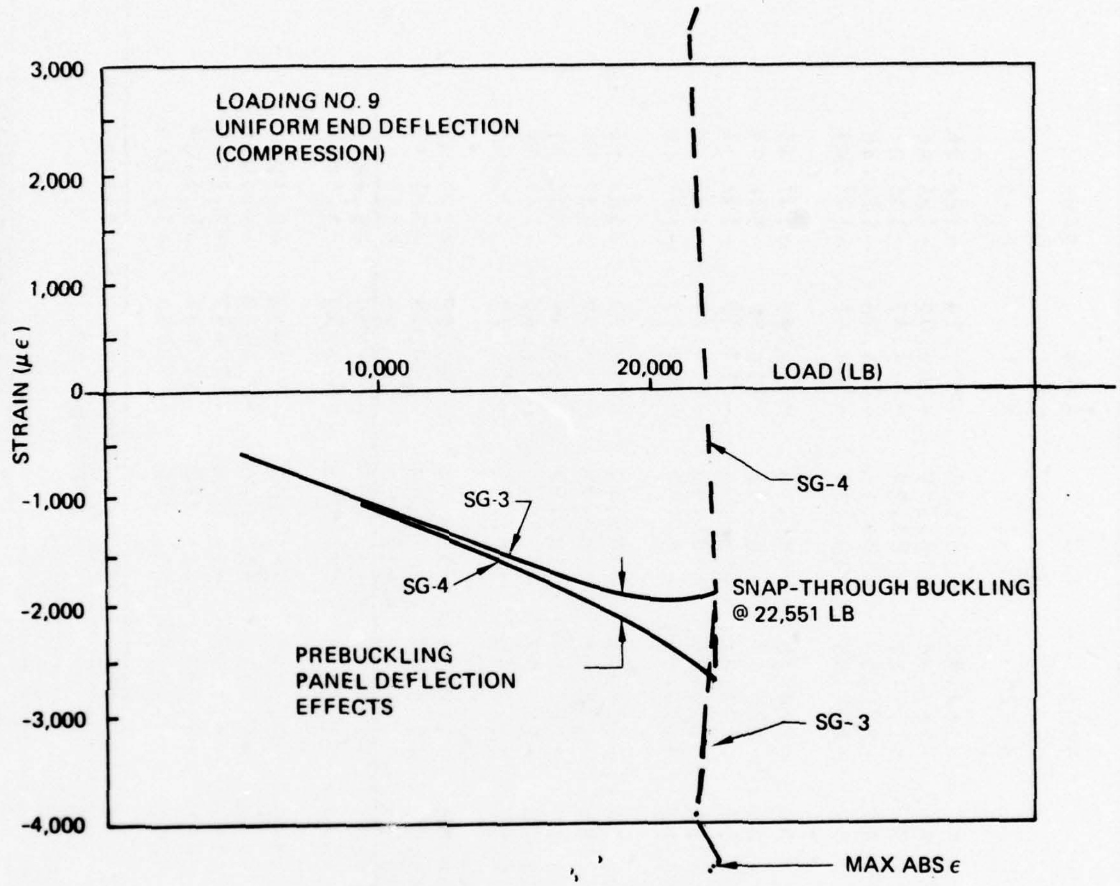

Figure 7-13. Subcomponent SC-2 Test Results

TIME IN DAY/HR/MIN/SEC	LOAD LBS X .01	SG-1 UE X 1	SG-2 UE X .1	SG-3 UE X .1	SG-4 UE X 1	SG-5 UE X .1
18/13 03/13	+186.81	-152.17	-164.88	-181.35	-200.14	-128.26
18/13 03/14	+188.49	-153.58	-167.24	-182.29	-202.02	-129.66
18/13 03/15	+191.00	-154.52	-169.12	-183.24	-204.37	-131.07
18/13 03/16	+192.68	-155.93	-171.00	-184.18	-206.26	-132.48
18/13 03/17	+194.37	-157.35	-172.88	-185.10	-208.61	-133.89
18/13 03/18	+196.05	-158.29	-174.76	-186.05	-210.48	-135.30
18/13 03/19	+196.89	-159.23	-176.17	-186.52	-211.89	-136.24
18/13 03/20	+197.73	-160.18	-177.57	-186.99	-213.30	-136.71
18/13 03/21	+198.58	-160.18	-178.04	-186.99	-213.77	-137.18
18/13 03/22	+198.58	-160.18	-178.51	-186.99	-213.77	-137.18
18/13 03/23	+198.58	-160.18	-178.51	-186.99	-213.77	-137.18
18/13 03/24	+198.58	-160.65	-178.51	-186.99	-213.77	-137.18
18/13 03/25	+197.73	-160.18	-178.51	-186.52	-213.77	-136.71
18/13 03/26	+196.05	-159.71	-177.57	-186.05	-211.89	-135.30
18/13 03/27	+191.84	-156.88	-172.88	-183.71	-206.73	-131.54
18/13 03/28	+186.81	-153.58	-168.18	-180.88	-201.08	-127.32
18/13 03/29	+181.75	-149.81	-163.48	-177.59	-194.98	-122.62
18/13 03/30	+174.17	-144.15	-155.49	-171.96	-186.06	-115.58
18/13 10/43 START	+3.3486	-1.4033	-1.8696	-8.0063	-6.1250	-1.8813
18/13 10/44	+3.3486	-1.4033	-1.8696	-8.0063	-6.1250	-1.8813
18/13 10/45	+4.1909	-1.8750	-2.3398	-8.9350	-7.0654	-2.8217
18/13 10/46	+6.7177	-3.2900	-4.2211	-11.286	-9.4169	-4.7026
18/13 10/47	+10.086	-5.1762	-6.1020	-14.578	-12.697	-7.0541
18/13 10/48	+12.614	-7.0625	-7.9833	-18.340	-15.989	-9.8754
18/13 10/49	+15.983	-9.4208	-10.334	-22.090	-19.750	-12.215

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Figure 7-14a.
Subcomponent No. 2 Strain Data For Loading No. 9

TIME IN DAY/HR/MIN/SEC	LOAD LBS X .01	SG-1 UF X .1	SG-2 UF X .1	SG-3 UF X .1	SG-4 UF X .1	SG-5 UF X .1
18/13 10/50	+19.352	-11.778	-12.674	-26.323	-23.501	-15.037
18/13 10/51	+22.701	-14.596	-15.966	-30.555	-27.733	-17.388
18/13 10/52	+26.913	-17.426	-18.787	-35.246	-31.966	-20.209
18/13 10/53	+30.282	-20.255	-21.609	-39.479	-36.187	-23.031
18/13 10/54	+33.651	-23.557	-24.419	-44.169	-40.889	-25.841
18/13 10/55	+37.863	-26.375	-27.710	-48.872	-45.122	-28.663
18/13 10/56	+42.074	-29.676	-31.002	-53.104	-49.813	-31.484
18/13 10/57	+46.265	-33.449	-34.765	-57.796	-54.045	-34.306
18/13 10/58	+49.635	-36.738	-38.045	-62.498	-58.736	-37.586
18/13 10/59	+53.846	-40.511	-41.807	-67.201	-63.439	-40.408
18/13 11/00	+58.058	-43.812	-45.099	-71.422	-67.671	-43.699
18/13 11/01	+62.270	-47.574	-48.849	-76.125	-72.362	-46.521
18/13 11/02	+66.460	-51.346	-52.611	-80.345	-76.595	-49.801
18/13 11/03	+70.672	-55.119	-56.374	-85.048	-81.286	-53.093
18/13 11/04	+75.726	-58.892	-60.124	-89.280	-85.518	-56.385
18/13 11/05	+79.938	-62.653	-63.886	-93.971	-90.221	-59.665
18/13 11/06	+84.149	-66.897	-68.119	-98.204	-94.912	-62.957
18/13 11/07	+89.183	-70.670	-71.869	-102.43	-99.144	-66.719
18/13 11/08	+93.394	-74.903	-76.102	-107.12	-103.84	-70.011
18/13 11/09	+98.448	-79.147	-80.334	-111.35	-108.53	-73.762
18/13 11/10	+102.66	-83.380	-84.555	-116.05	-113.71	-77.523
18/13 11/11	+107.69	-87.625	-88.787	-120.75	-118.40	-81.286
18/13 11/12	+112.74	-92.341	-93.489	-125.45	-123.57	-85.036
18/13 11/13	+117.80	-96.574	-97.710	-130.14	-128.73	-88.708
18/13 11/14	+122.85	-100.81	-102.41	-134.37	-134.37	-93.031

Figure 7-14b.
Subcomponent No. 2 Strain Data For Loading No. 9

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TIME IN DAY/HR/MIN/SEC	LOAD LBS X .01	SG-1 UE X .1	SG-2 UE X .1	SG-3 UE X .1	SG-4 UE X .1	SG-5 UE X .1
16/13 11/15	+127.88	-105.53	-107.10	-139.54	-139.55	-96.781
16/13 11/16	+132.94	-110.23	-111.80	-143.77	-145.18	-101.01
16/13 11/17	+138.83	-114.95	-116.98	-148.94	-150.82	-105.24
16/13 11/18	+143.89	-119.65	-121.67	-153.63	-156.45	-109.46
16/13 11/19	+149.76	-124.37	-126.84	-157.86	-162.57	-113.69
16/13 11/20	+154.82	-129.09	-132.00	-162.56	-168.67	-118.39
16/13 11/21	+160.71	-133.79	-137.64	-167.26	-174.78	-122.62
16/13 11/22	+166.61	-138.51	-142.80	-171.49	-180.88	-127.32
16/13 11/23	+171.64	-143.21	-148.45	-175.71	-187.46	-131.54
16/13 11/24	+177.54	-147.93	-154.08	-179.94	-194.04	-136.24
16/13 11/25	+183.44	-152.64	-159.72	-183.71	-200.61	-140.93
16/13 11/26	+188.49	-157.35	-165.35	-186.99	-207.67	-145.17
16/13 11/27	+194.37	-162.06	-171.47	-190.28	-214.71	-149.87
16/13 11/28	+199.42	-166.30	-177.57	-193.10	-222.22	-154.56
16/13 11/29	+205.32	-170.54	-183.21	-194.98	-230.22	-158.79
16/13 11/30	+210.37	-174.31	-189.31	-196.39	-238.67	-163.50
16/13 11/31	+215.40	-178.08	-195.43	-196.39	-247.59	-168.19
16/13 11/32	+220.46	-181.38	-201.53	-194.04	-257.93	-172.42
16/13 11/33	+225.51	-184.68	-207.64	-187.46	-270.61	-177.11
16/13 11/34	+219.61	-180.43	-206.70	-262.16	+162.53	-188.39
16/13 11/35	+217.93	-176.20	-204.82	-397.93	+330.71	-191.68
16/13 11/36	+220.46	-175.73	-207.64	-416.25	+356.55	-197.32
16/13 11/37	+223.83	-175.73	-210.94	-427.06	+374.88	-202.05
16/13 11/38	+225.51	-175.73	-214.69	-435.98	+380.44	-208.12
16/13 11/39 END	+224.67	-174.79	-213.27	-436.92	+391.32	-207.65

Figure 7-14c.
Subcomponent No. 2 Strain Data For Loading No. 9

CALC	06/19/75	MFW	REVISED	DATE	PRIMARY AIRCRAFT STRUCTURE SPECIMEN NO. 2	SLC22
CHECK						
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APR						PAGE
CONTRACT NO.						

TIME IN DAY/HR/MIN/SEC	LOAD LBS X .01	SG-6 UE	SG-7 UE X .1	SG-8 UE X .1	SG-9 UE X .1
18/13/03/13	+186.81	-65.649	-186.46	-160.22	-171.48
18/13/03/14	+188.49	-60.960	-188.32	-161.63	-174.30
18/13/03/15	+191.00	-60.960	-190.28	-163.51	-176.64
18/13/03/16	+192.68	-60.960	-192.07	-164.91	-178.99
18/13/03/17	+194.37	-60.960	-194.42	-166.32	-180.87
18/13/03/18	+196.05	-60.960	-195.82	-167.73	-183.22
18/13/03/19	+196.89	-56.271	-197.70	-169.14	-184.63
18/13/03/20	+197.73	-56.271	-198.64	-169.61	-185.58
18/13/03/21	+198.58	-56.271	-199.09	-170.08	-186.05
18/13/03/22	+198.58	-56.271	-199.09	-170.08	-186.05
18/13/03/23	+198.58	-56.271	-199.09	-170.55	-186.05
18/13/03/24	+198.58	-56.271	-199.56	-170.55	-186.05
18/13/03/25	+197.73	-56.271	-199.09	-170.08	-185.58
18/13/03/26	+196.05	-60.960	-197.70	-168.67	-183.22
18/13/03/27	+191.84	-65.649	-193.01	-165.38	-176.64
18/13/03/28	+186.81	-70.338	-188.32	-161.63	-169.13
18/13/03/29	+181.75	-75.027	-183.17	-157.39	-161.62
18/13/03/30	+174.17	-79.717	-175.21	-150.35	-149.40
18/13/10/43 START	+3.3486	-14.067	-7.5029	-4.2441	-1.8813
18/13/10/44	+3.3486	-14.067	-7.5029	-4.2441	-2.3515
18/13/10/45	+4.1909	-14.067	-8.9096	-5.1845	-2.8217
18/13/10/46	+6.7177	-14.067	-11.254	-6.5952	-4.7026
18/13/10/47	+10.086	-14.067	-14.525	-8.9467	-6.1137
18/13/10/48	+12.614	-9.3789	-17.338	-11.286	-7.9946
18/13/10/49	+15.983	-9.3789	-20.821	-14.107	-9.8754

Figure 7-14d.
Subcomponent No. 2 Strain Data For Loading No. 9

CALC	06/19/75	MFW	REVISED	DATE	PRIMARY AIRCRAFT STRUCTURE SPECIMEN NO. 2	SL022
CHECK						
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CONTRACT NO.						

TIME IN DAY/HR/MIN/SEC	LOAD LBS X .01	SG-6 UF	SG-7 UF X .1	SG-8 UF X .1	SG-9 UF X .1
18/13 10/50	+19.352	-9.3789	-23.892	-16.929	-11.745
18/13 10/51	+22.701	-9.3789	-27.644	-19.750	-14.096
18/13 10/52	+26.913	-9.3789	-30.926	-22.572	-16.447
18/13 10/53	+30.282	-9.3789	-34.208	-25.853	-18.799
18/13 10/54	+33.651	-9.3789	-37.949	-29.145	-21.150
18/13 10/55	+37.863	-9.3789	-41.231	-31.966	-23.960
18/13 10/56	+42.074	-9.3789	-44.982	-35.716	-26.782
18/13 10/57	+46.265	-9.3789	-48.722	-39.008	-30.073
18/13 10/58	+49.635	-9.3789	-52.474	-42.300	-33.835
18/13 10/59	+53.846	-9.3789	-56.225	-46.062	-37.586
18/13 11/00	+58.058	-14.067	-59.965	-49.813	-41.818
18/13 11/01	+62.270	-14.067	-64.186	-53.104	-45.581
18/13 11/02	+66.460	-14.067	-67.937	-56.867	-49.801
18/13 11/03	+70.672	-14.067	-71.677	-60.617	-54.504
18/13 11/04	+75.726	-14.067	-75.897	-64.379	-59.195
18/13 11/05	+79.938	-14.067	-80.118	-68.142	-64.368
18/13 11/06	+84.149	-14.067	-84.327	-71.892	-70.011
18/13 11/07	+89.183	-9.3789	-88.547	-75.654	-75.172
18/13 11/08	+93.394	-9.3789	-92.767	-79.416	-80.815
18/13 11/09	+98.448	-9.3789	-96.976	-83.637	-86.447
18/13 11/10	+102.66	-9.3789	-101.66	-87.869	-92.090
18/13 11/11	+107.69	-9.3789	-106.34	-91.631	-98.192
18/13 11/12	+112.74	-9.3789	-110.56	-95.852	-103.83
18/13 11/13	+117.80	-9.3789	-115.72	-100.08	-109.93
18/13 11/14	+122.85	-4.6894	-120.39	-104.78	-116.05

Figure 7-14e.
Subcomponent No. 2 Strain Data For Loading No. 9

CALC	06/19/75	MFW	REVISED	DATE	PRIMARY AIRCRAFT STRUCTURE SPECIMEN NO. 2	SLC22
CHECK						
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APR					THE BOEING COMPANY	PAGE
CONTRACT NO.						

TIME IN DAY/HR/MIN/SEC	LOAD LBS X .01	SG-6 UF	SG-7 UF X .1	SG-8 UF X .1	SG-9 UF X .1
18/13/11/15	+127.88	-4.6894	-125.08	-109.00	-122.62
18/13/11/16	+132.94	-4.6894	-130.23	-113.71	-129.19
18/13/11/17	+138.83	+0.0000	-135.39	-117.93	-135.77
18/13/11/18	+143.89	+0.0000	-140.54	-122.63	-142.35
18/13/11/19	+149.76	+0.0000	-146.16	-127.33	-149.40
18/13/11/20	+154.82	+0.0000	-151.32	-132.02	-156.44
18/13/11/21	+160.71	+4.5747	-156.94	-136.73	-163.50
18/13/11/22	+166.61	+4.5747	-162.56	-141.42	-171.01
18/13/11/23	+171.64	+9.2636	-168.18	-146.12	-178.52
18/13/11/24	+177.54	+9.2636	-174.28	-150.82	-186.05
18/13/11/25	+183.44	+13.953	-179.89	-155.51	-194.03
18/13/11/26	+188.49	+18.642	-185.52	-160.22	-201.54
18/13/11/27	+194.37	+23.331	-191.60	-165.38	-209.54
18/13/11/28	+199.42	+28.021	-197.70	-169.61	-217.52
18/13/11/29	+205.32	+32.709	-203.31	-174.31	-225.50
18/13/11/30	+210.37	+37.399	-209.41	-179.00	-233.97
18/13/11/31	+215.40	+42.088	-215.50	-183.24	-241.95
18/13/11/32	+220.46	+51.466	-221.59	-187.46	-249.93
18/13/11/33	+225.51	+56.156	-227.68	-191.22	-257.93
18/13/11/34	+219.61	+107.73	-230.49	-172.43	-275.30
18/13/11/35	+217.93	+103.04	-230.02	-158.33	-285.17
18/13/11/36	+220.46	+121.69	-233.77	-156.92	-294.56
18/13/11/37	+223.83	+140.44	-237.98	-156.92	-303.50
18/13/11/38	+225.51	+159.20	-241.73	-156.45	-309.60
18/13/11/39 END	+224.67	+159.20	-240.33	-155.04	-308.19

Figure 7-14f.
Subcomponent No. 2 Strain Data For Loading No. 9

CALC	06/19/75	MFW	REVISED	DATE	PRIMARY AIRCRAFT STRUCTURE SPECIMEN NO. 2	SLC22
CHECK						
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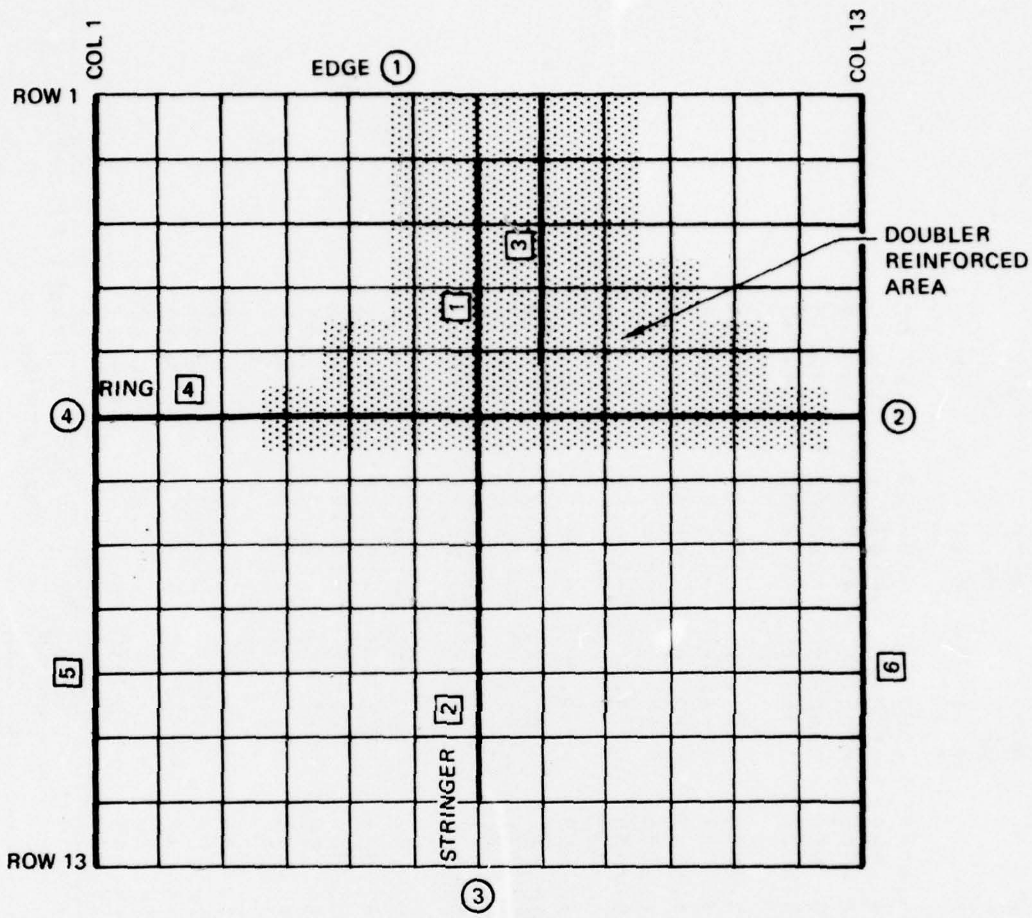
CONTRACT NO.						

2-2600
D3 5568 1750 ORIG. 3/71



L = 19.5
R = 12.50
$\Theta = 90^{o}$
W = 19.635

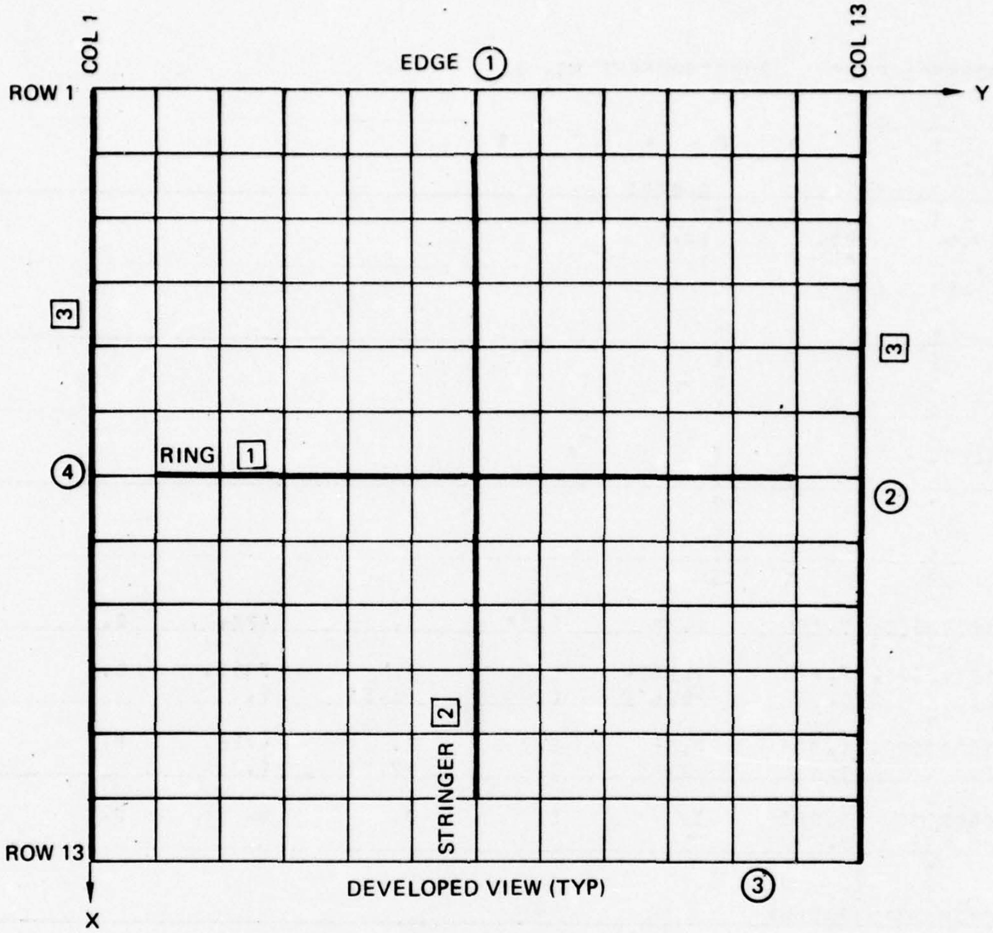
EDGE	DISPLACEMENT BOUNDARY CONDITIONS			
	U	V	W	β
1	1	0	0	0
2	1	1	0	1
3	0	0	0	1
4	1	1	0	0

0 = FIXED
1 = FREE

Figure 7-15a. Stags–B Model of Subcomponent No. 1

```
FIRESC1 FILE     SUBCOMPONENT NO. 1
    1     1     1
   13    13
    1     1     4     0     4     0     3
1.0
    3.   20 0.          0.0001
    1     1     0     0
19.5        90.         12.5
0.          0.          0.
   13    13
    0.
    1     0     0     0
    1     1     0     1
    0     0     0     0
    1     1     0     1
    1     0
1000.             1     1     1     0
    6     1     1    13
    7     2     2    12
    8     3     1     6
    1     4     1    13
   13     4     1    13
    0
10000000. 0.75       0.28       0.28       0.         42847.     0.         -.35
    3
10000000. 0.628      0.1654     0.         0.         2363.      0.         -0.682
0.        0.         -0.682     0.         -1.58      0.
    3
10000000. 1.875      0.65       2.0        0.         4726.      0.         -0.65
0.        0.         -0.65      0.         -2.08      0.
    1
6640000.  0.163      0.         0.         0.         0.         0.         -0.088
-0.088    0.
    1     1     1
    1     0     0
SKIN WITH PAD-UP
    2     0
    1
666872.   197128.    477398.    232364.
495.6     117.1      297.5      140.
1333744.  394256.    954796.    464728.
3573.5    997.5      2443.6     1179.6
PAD-UP AT 33 NODES
   33
  1 6  1 7  1 8  1 9  2 6
  2 7  2 8  2 9  3 6  3 7
  3 8  3 9  4 6  4 7  4 8
  4 9  410  5 5  5 6  5 7
  5 8  5 9  510  511  6 4
  6 5  6 6  5 7  6 8  6 9
  610  611  612
19.5        90.
```

Figure 7-15b. STAGS-B Model of Subcomponent No. 1



L = 18.0
R = 12.45
$\Theta = 82.84^{\circ}$
W = 18.0

EDGE	DISPLACEMENT BOUNDARY CONDITIONS			
	U	V	W	β
1	1	0	0	0
2	1	1	0	1
3	0	0	0	1
4	1	1	0	0

0 = FIXED
1 = FREE

Figure 7-16a. Stags–B Model of Subcomponent No. 2

```
FIRESC2 FILE      SUBCOMPONENT NO. 2
     1      1      1
    13     13
     1      1      3      3      3      1      3
1.0
     2     20 0.               0.0001
     1      1      0      3
18.        62.04           12.45
0.          0.              0.
    13     13
     0
     1      0      1      0
     1      1      0      1
     0      0      0      0
     1      1      0      1
     1
0.016             -1      1      1      1
     7      1      2     12
     7      2      2     12
     1      3      1     13
    13      3      1     13
     2
66-0000.   0.26         0.1575     0.         0.          1169.1     0.          -0.66
0.         0.           -1.6       0.
     3
100.0000.  0.3371       0.06.6     0.         0.          2363.      0.          -0.726
0.          .           -0.726      .         -1.56       0.
     1                 3
.6640000.  0.035        0.         0.         0.          0.         0.          -0.066
-0.066     0.
     1      1      1
     1      0
UNIFORM WALL
     1
     1
564593.    166323.      41-217.    166-99.
346.4      51.          2.5.6      2.2.9
16.        62.64
```

Figure 7-16b. STAGS-B Model of Subcomponent No. 2

The linear buckling analysis results from the STAGS-B models were:

SUBCOMPONENT	STAGS-B BUCKLING SOLUTION	ACTUAL TEST VALUE
SC-1	P_{cr} = 36508 lb	44682 lb
SC-2	End Δx = 0.0538 in.	0.0515 in.

The subcomponent No. 1 model results are lower than test apparently because of the assumption of simple edge supports along the reaction edge. The subcomponent No. 1 model had fixed loaded and reaction edge rotations and the critical end deflection agrees closely with the actual load-deflection results. Based on these subcomponent No. 2 results, and the tolerance to prebuckling deformations demonstrated by the testing, the use of a large "knock-down" factor on theoretical buckling calculations was assumed to be unnecessary in the structural analysis of the delivered components (see Section 9.0).

From a design point-of-view, the subcomponent No. 1 test results allowed a lighter design which was reflected in the subcomponent No. 2 details. Based on the subcomponent No. 2 test results, the final component designs were lightened further with respect to stringer area and deletion of the SGF/Ps skin filler plies. This was possible because of the small panel sizes present in the component designs; a separate STAGS-B analysis indicated the final stringers had sufficient stiffness to preclude a general instability failure mode.

8.0 COMPONENT FABRICATION

The fabrication activities associated with the delivered components are highlighted and summarized in this section.

8.1 FABRICATION PROCEDURES

Figure 8-1 is the sequence of fabrication steps that were followed with respect to processes. Materials that were used are specified on the detail drawing SK-031275KO in Appendix A. Unless specifically noted, the materials and processes used for the components are the same as used for subcomponent No. 2 and are discussed in the Fabrication Processes Section (Section 5.0).

The tool used for post-forming and fusing skin laminates is shown in Figure 8-2. Roll-formed stainless steel (304 alloy, 0.125 in. gage) was used to preclude corrosion problems. The tool surface was covered with a Kapton film sprayed with Frekote release agent which separated the laminates from the tool. Threaded clevis rods connected the ends of the tool to allow precise radius adjustment. After the skins were fused, the tool was employed for ring forming and final component assembly bonding.

Figure 8-3 shows a typical vacuum bag setup used for skin laminate fusing. The reuseable silicone rubber bag sheet is peeled back to reveal the glass bleeder plies, silicone rubber edge sealant, and a skin laminate. Figure 8-4 is the interior side of a skin after fusing; the matte appearance is due to imprint from the glass bleeder fabric.

The stringer fusing was accomplished using the tooling and laminate preforms shown in Figure 8-8. Figure 8-9 is a completed stringer prior to final trimming.

The components were trimmed to final dimensions after adhesive bonding of the skin-stiffener assemblies. Trimming was done with conventional handheld saber saws and mylar templates. The skin laminates were remarkably

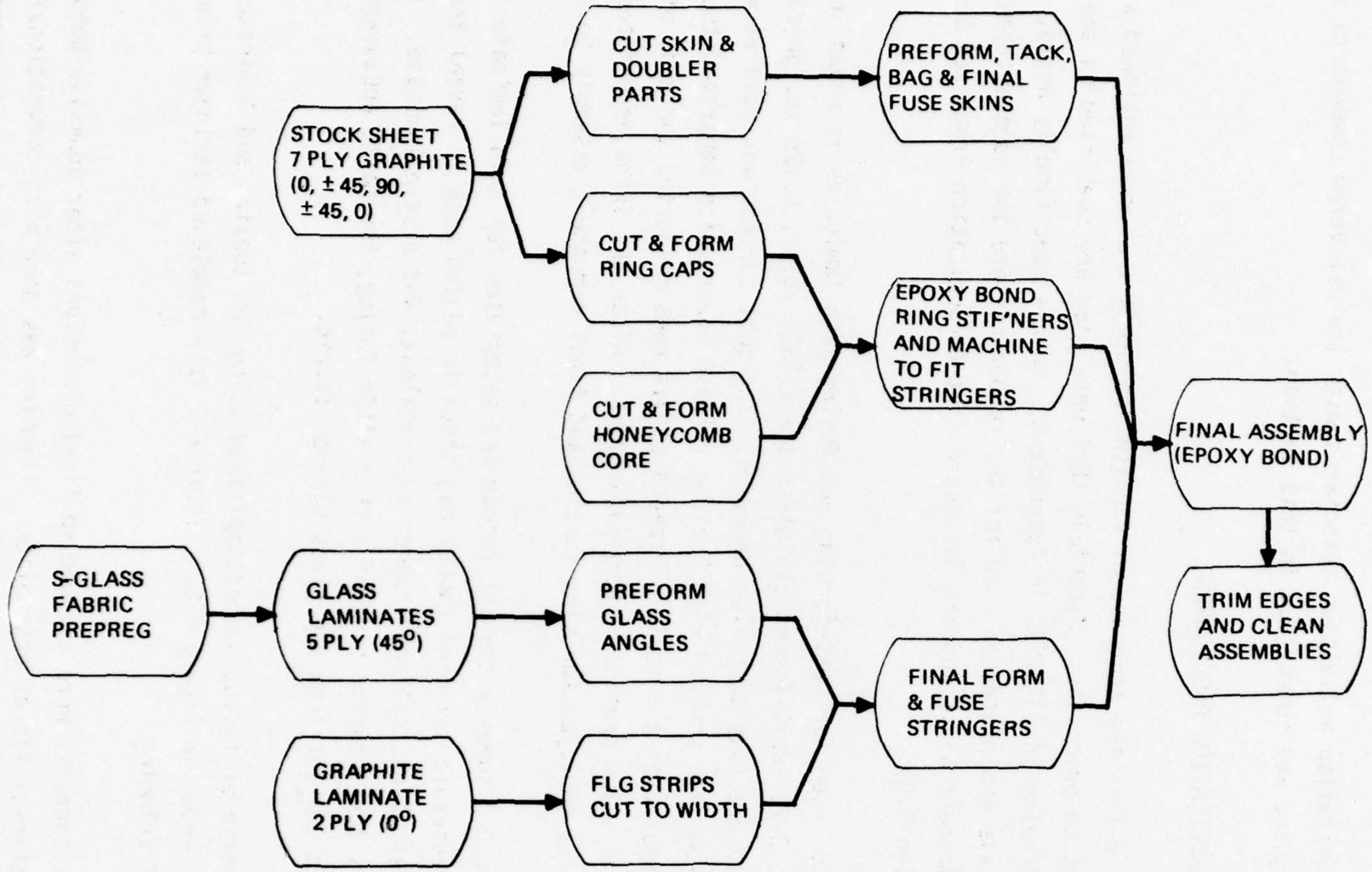


Figure 8-1. Component Fabrication Operations

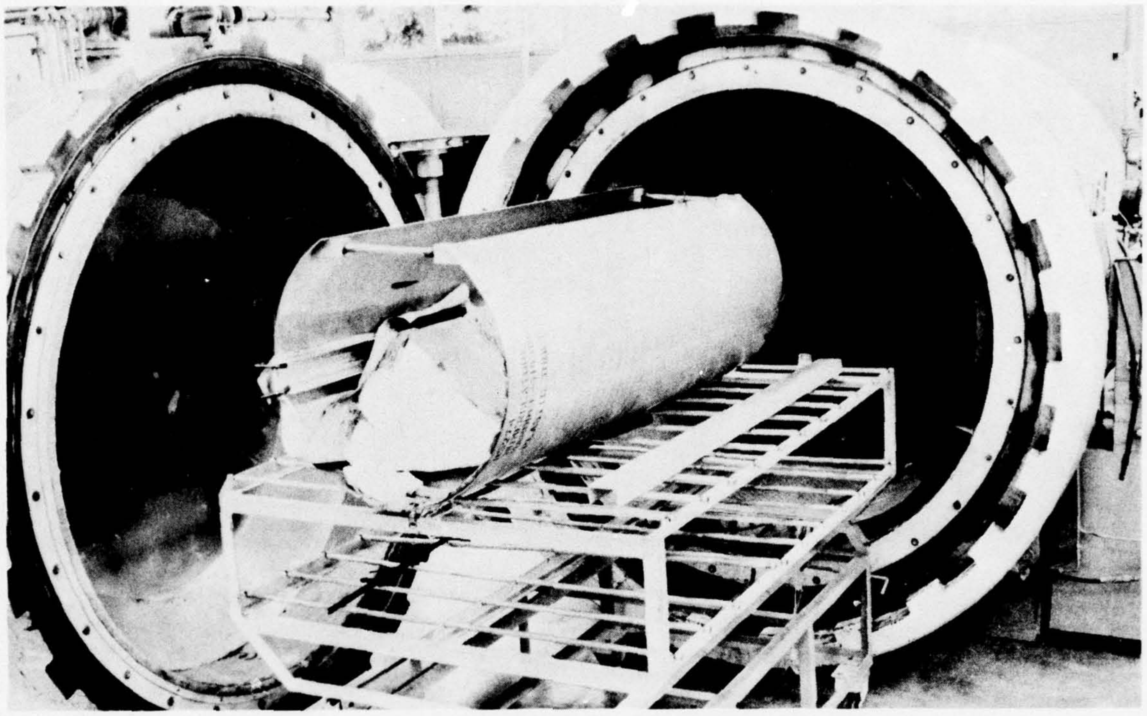


Figure 8-2. Post-Forming and Fusing Tool for Component Skins

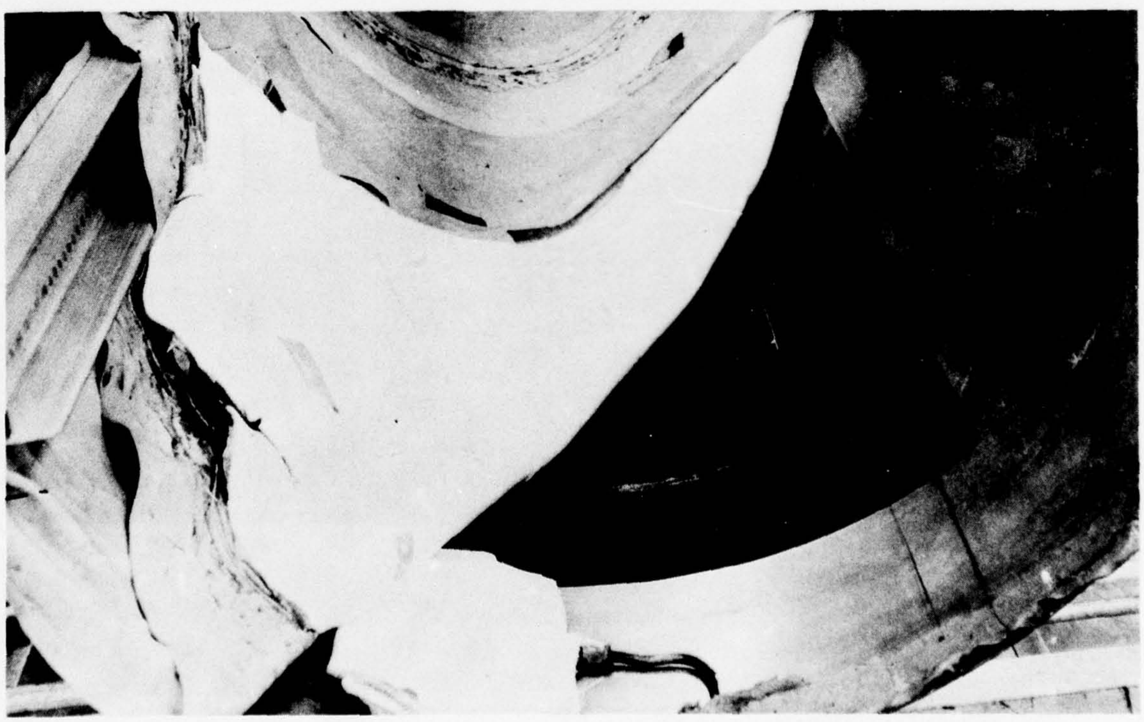


Figure 8-3. Skin Fusing Silicone Rubber Vacuum Bag Set-Up

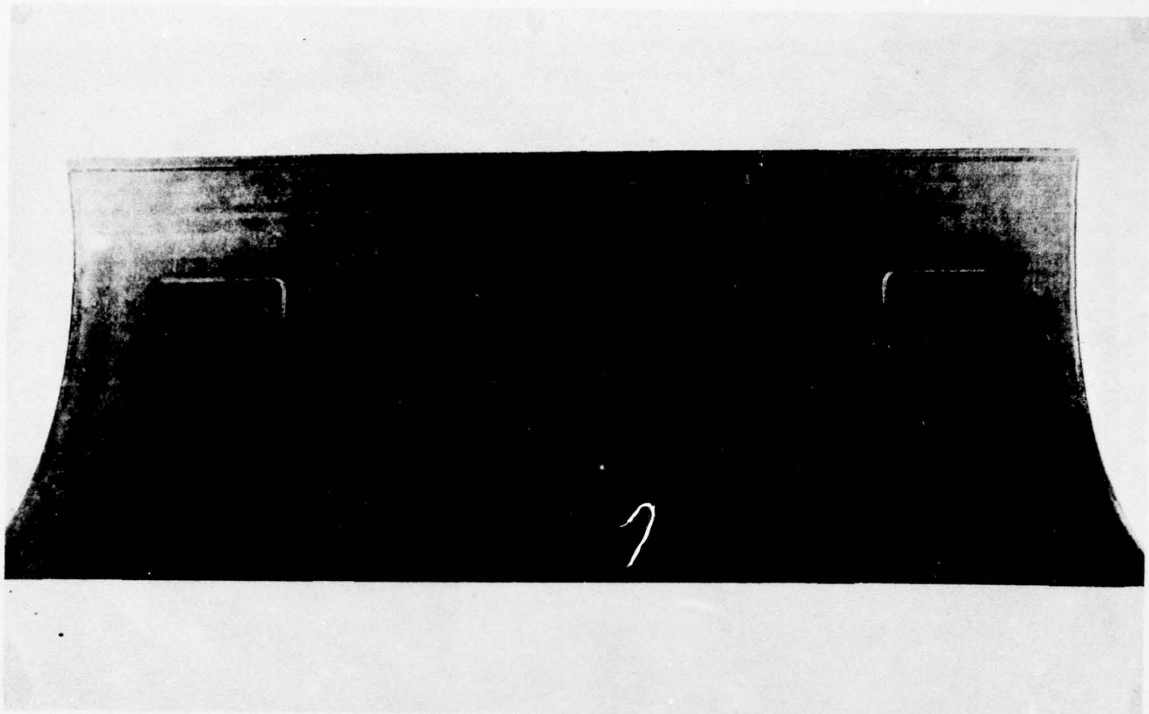


Figure 8-4. Left Skin After Fusing

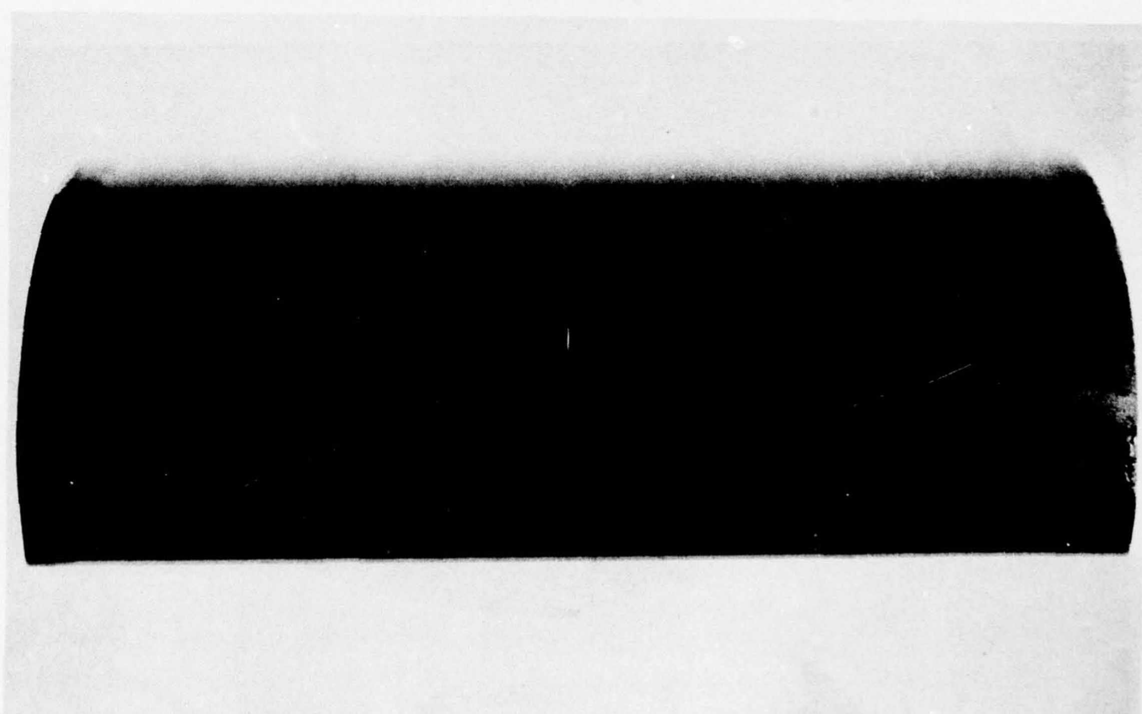


Figure 8-5. Dimples From Wrinkled Kapton Parting Film in Left Skin Surface

tough and no special procedures or fine tooth blades were required. There was no evidence of edge delamination. Areas such as the fuel fill port were cut successfully despite difficulty in overcoming chatter of the hand-held saw. Edge dressing to final dimensions was accomplished rapidly by hand with sanding blocks.

8.2 SKIN REWORKING

One problem was encountered in fusing the skin for the left side component. Figure 8-5 shows the skin laminate which has a barely discernible dimple with the form of white hair-like lines. This defect was due to a wrinkle in the Kapton tool release film which left an impression in the laminate. Figure 8-6 is a close-up photograph of a dimpled area. The skin part was successfully reworked by performing the fusing process over again. Figure 8-7 shows the reworked skin with the defects completely removed; this reworking experience was an interesting demonstration of the possible benefits of using thermoplastic (recyclable) composites. Had this been an epoxy component, the entire part would be rejected due to the defect.

8.3 NON-DESTRUCTIVE INSPECTION

Ultransonic inspection of the left component was performed in a related Boeing IRAD program; the laboratory set-up shown in Figure 8-10 was used for C-scanning. Figure 8-11 is a C-scan signature of a stringer area in which the skin doubler and stringer details are clearly visible. Excess resin beads are also visible. No defects were detected by the C-scan inspection and by tapping inspection of the left side component. The two other components were inspected visually and by tapping were judged to be free of defects.

8.4 INSTALLATION ON FIREBEE FUSELAGE

The following procedure steps were followed by NAVAIRDEVCEN to install the Gr/Ps components on the test XBQM-34E center body section:

Figure 8-6. Close-Up of Skin Dimples in Left Skin

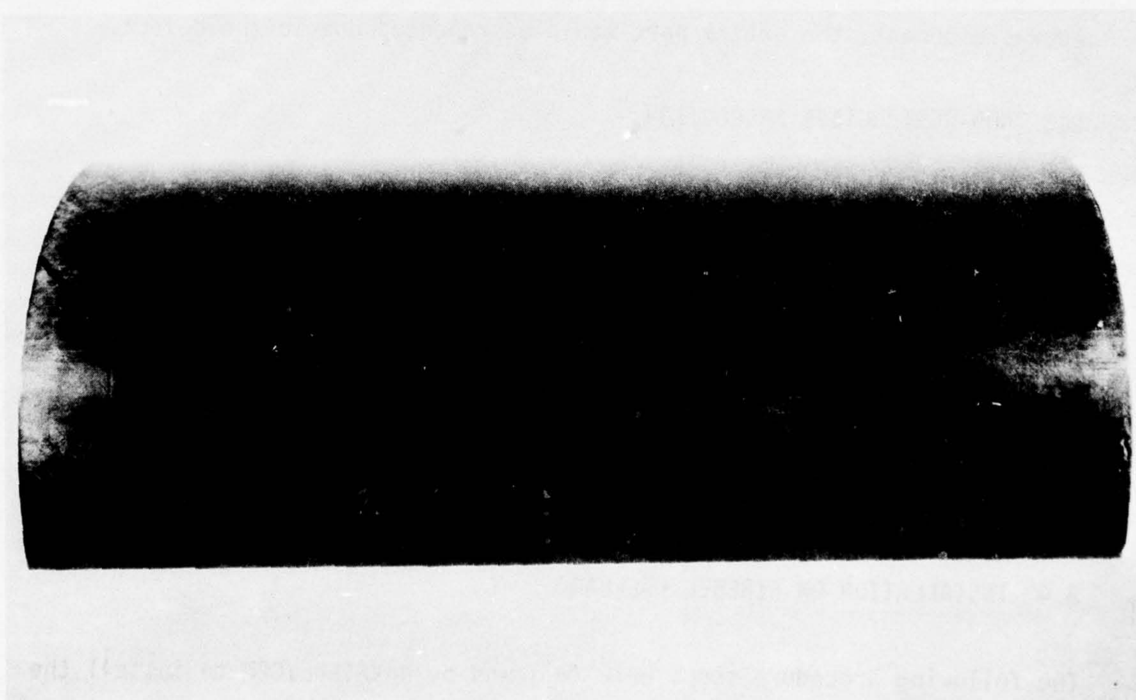


Figure 8-7. Left Skin After Re-Fusing With Dimples Completely Removed

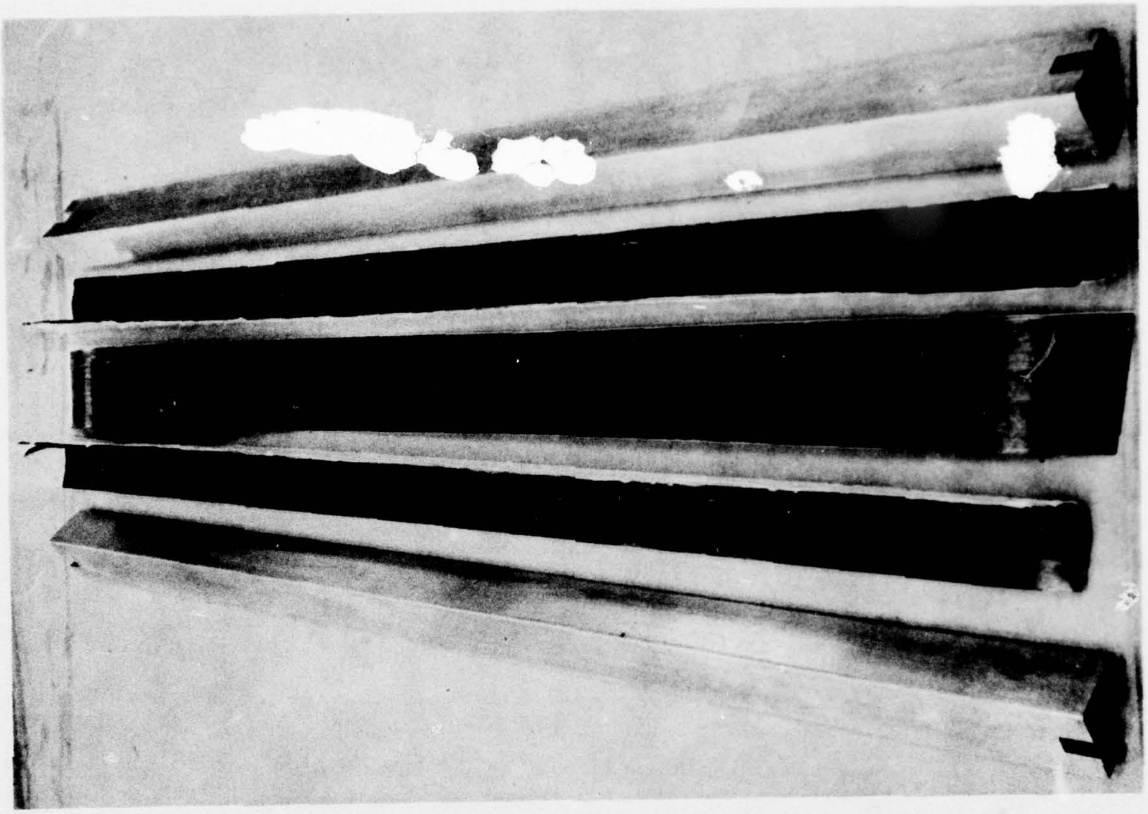


Figure 8-8. Component Stringer Parts and Tooling Prior to Fusing

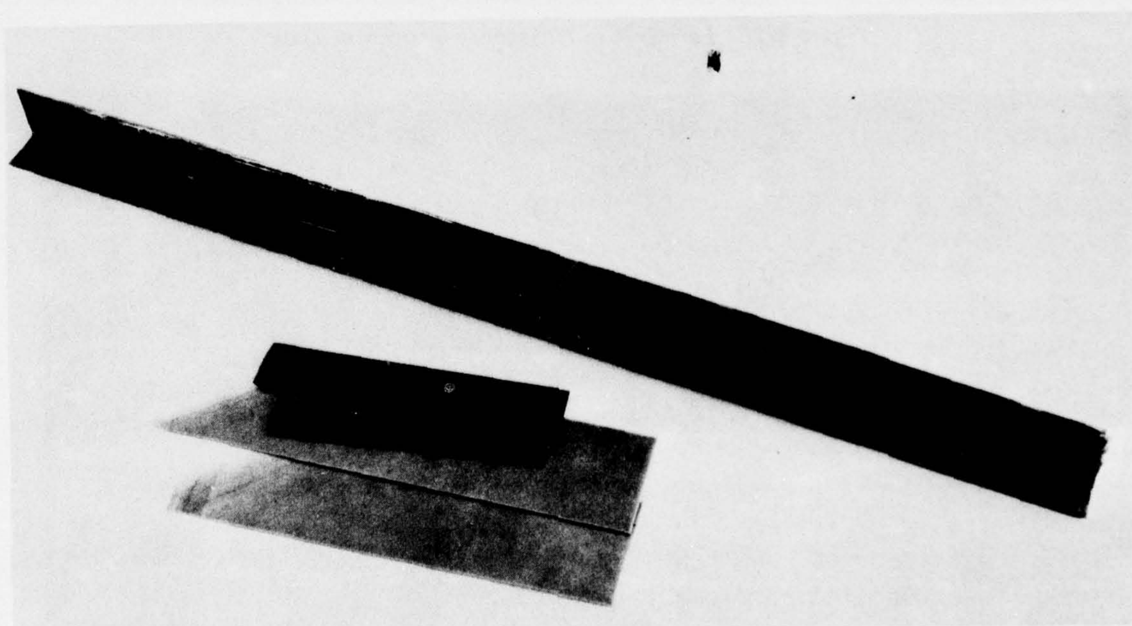


Figure 8-9. Fused Stringer Prior to Trimming

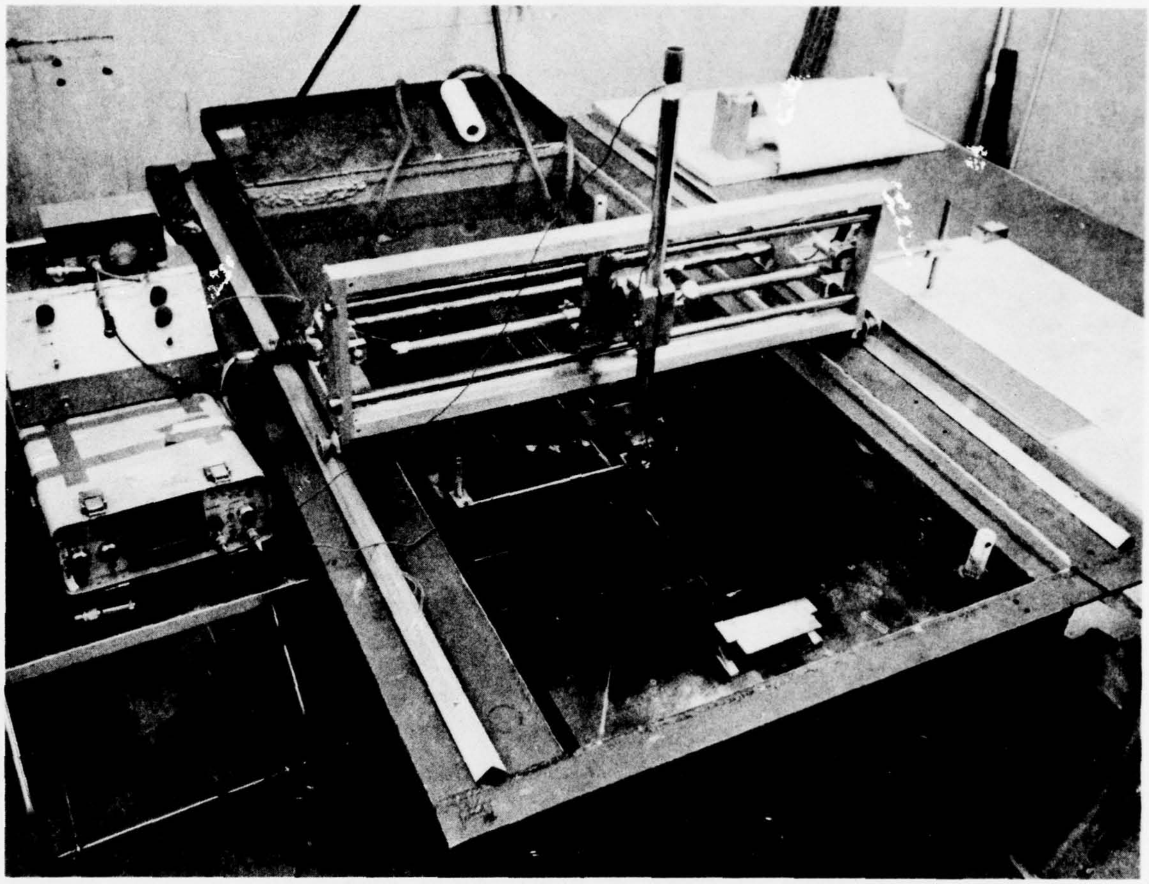


Figure 8-10. Left Skin in Ultrasonic Inspection Tank

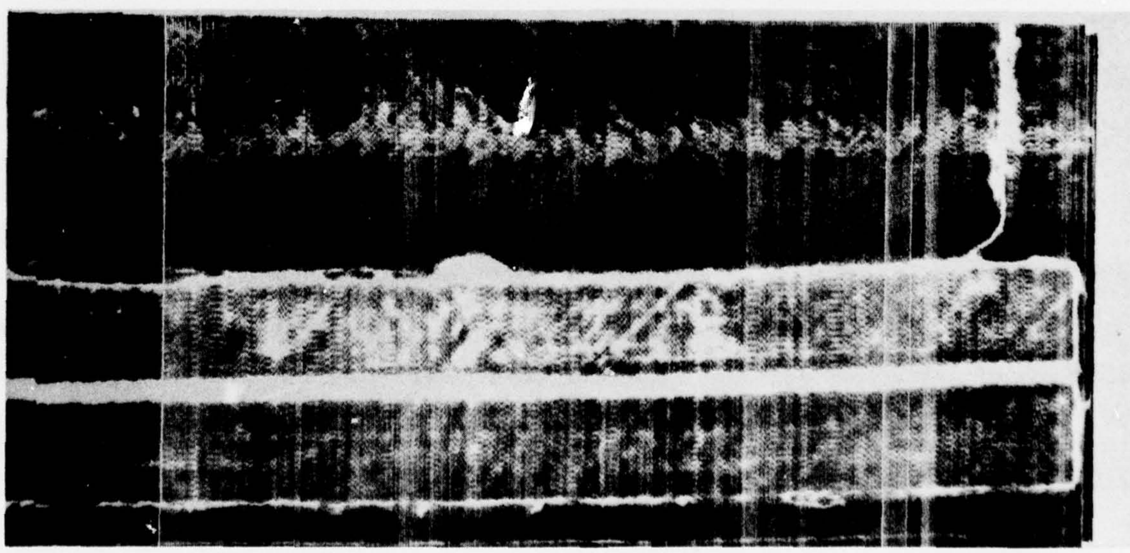


Figure 8-11. Ultrasonic Scan of Stringer Area on Left Skin

1. Remove aluminum skin parts and cut existing stringers and ring frames per Drawing SK 30675JL (see Appendix A).

2. Clean-up all metal faying surfaces per Structural Repair Manual (SRM), NAVAIR 01-100TBB-3.

3. Prepare drilling template for components.

4. Drill components (carbide drills, undersize holes for 3/16 Huck blind bolts).

5. Fit components and ream fastener holes (carbide tools).

6. Countersink holes (carbide tools).

7. Fabricate metal airframe fittings per Drawings SK 30675JL and SK 031275K).

8. Clean parts per SRM.

9. Install components per SRM.

10. Connect strain gage leads.

Installation of the components was accomplished using methods and tools routinely used for metal and glass-reinforced plastic fabrication. The installed components are shown in Figures 8-12 and 8-13.

The only major problem that was encountered was in fit-up of the side Gr/Ps components to the existing bulkheads; the new skins had inside radii that were smaller than the nominal design value. Proper fit-up was achieved by cutting the ring-stiffener flanges just above each central stringer. This allowed the skins to flex sufficiently during final fit-up to the bulkheads. The error in the component curvature is attributed to improper stiffener bonding and/or to relaxation of the skins during the lengthy

interval between stiffener bonding and installation. Because the ring stiffeners were cut in an area of low load, the final fit-up was achieved without affecting structural performance.

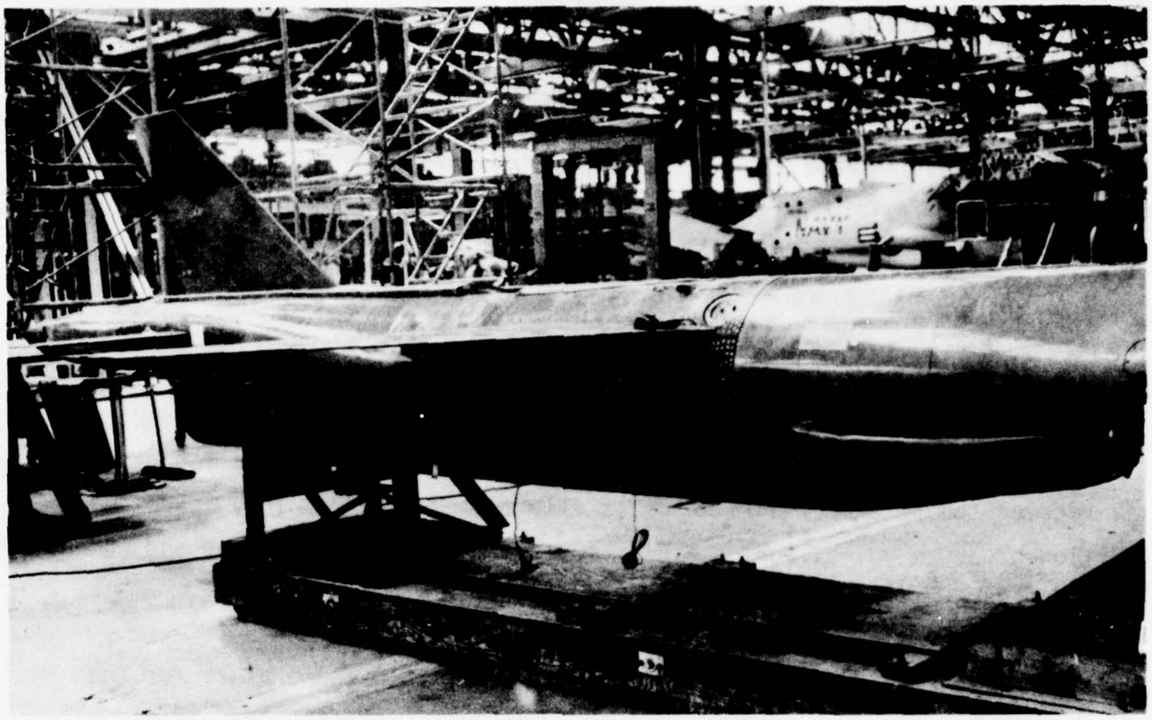


Figure 8-12: XBQM-34E With Installed Gr/Ps Centerbody Components

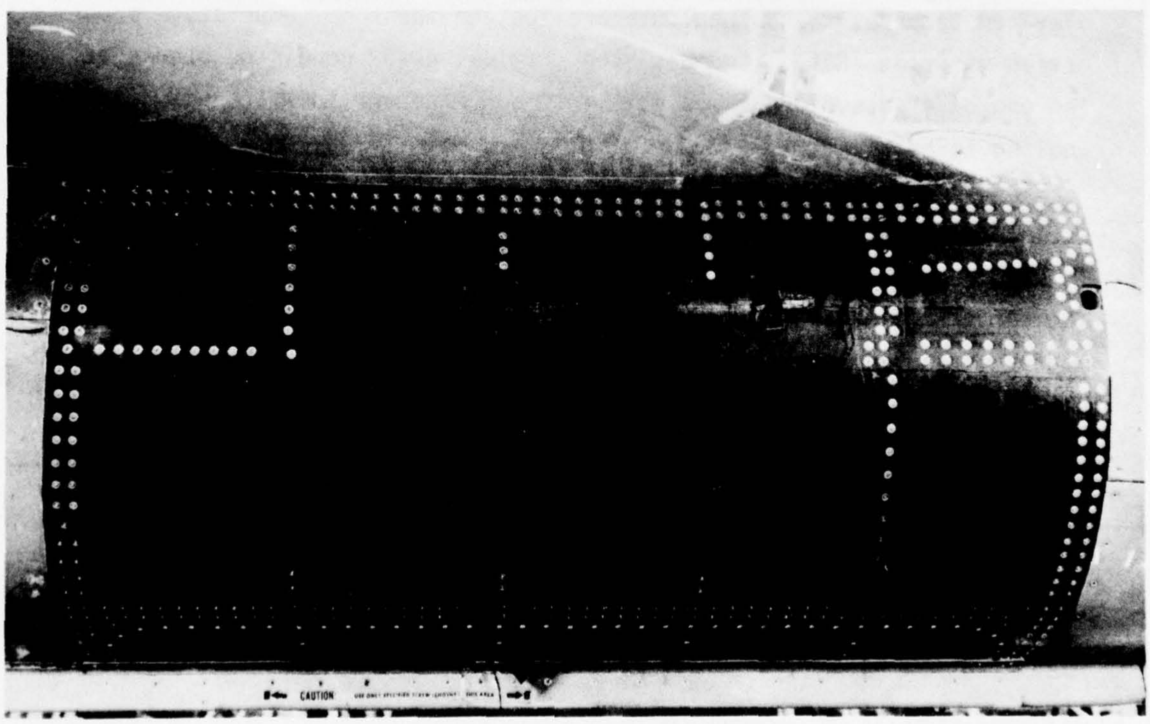


Figure 8-13: Installed Gr/Ps Left Skin Component

9.0 COMPONENT STRUCTURAL ANALYSES

This section presents the structural analysis results for the delivered components.

9.1 DESIGN CONDITIONS

The design conditions for the production BQM-34E centerbody section are shown in Figure 9-1 (References 6, 8 and 9). The critical load condition relative to skin sizing is 4PX02 (Teledyne Ryan's designation for main recovery chute deployment) which is illustrated in 9-2. This load condition produces point loading at the forward main chute riser lug and moments and shears at the ends of the centerbody section.

The fuselage skin temperature is assumed to be room temperature for the structural analyses. The maximum pre-ducted skin surface radiation equilibrium temperature is 186°F during on-the-deck dash at Mach 1.2; this elevated temperature has negligible effect on the room temperature properties of Gr/Ps (Ref. 1 and 2). The critical 4PX02 condition occurs at the end of a subsonic pull-up fuel jettison maneuver in which structural cooling takes place.

9.2 FINITE ELEMENT LOADS ANALYSIS

The centerbody section internal loads were determined with a finite element model using Boeing's SAMECS finite element analysis code. The modeling was necessary because of complex load paths resulting from (1) the half-to-full circular body transition at sta. 233.5 and (2) the slot in the skin for the wing carry-through. The model features are shown in Figure 9-3. The baseline aluminum skin was modeled; this was satisfactory for loads analysis because of the closeness of membrane stiffness with the composite design. The maximum skin load conditions for the 4PX02 condition are given in Figure 94 and were used in the panel buckling analyses.

1. Main recovery chute deployment load
   Ryan 4PX02 condition

2. Fuel pressure

Pressurization system	5.5 PSIG limit
Ground launch induced pressure head @ ring sta. 258.34	5.5 PSIG limit
Total	11.0 PSIG limit
	(17.0 PSIG ult)

3. Wing mount bolt loads due to flight maneuver

Bolt location	Load
1 (Fwd)	-331.2 lb ult
2	-277.0
3	106.7
4	2163.5
5	3204.7

4. External fuel tank load at explosive bolt attachment sta 258.34 for 5G limit pull-up flight condition
   1,000 lb bolt pre-load + 2,293 lb limit down bolt load

Figure 9-1. Design Load Conditions

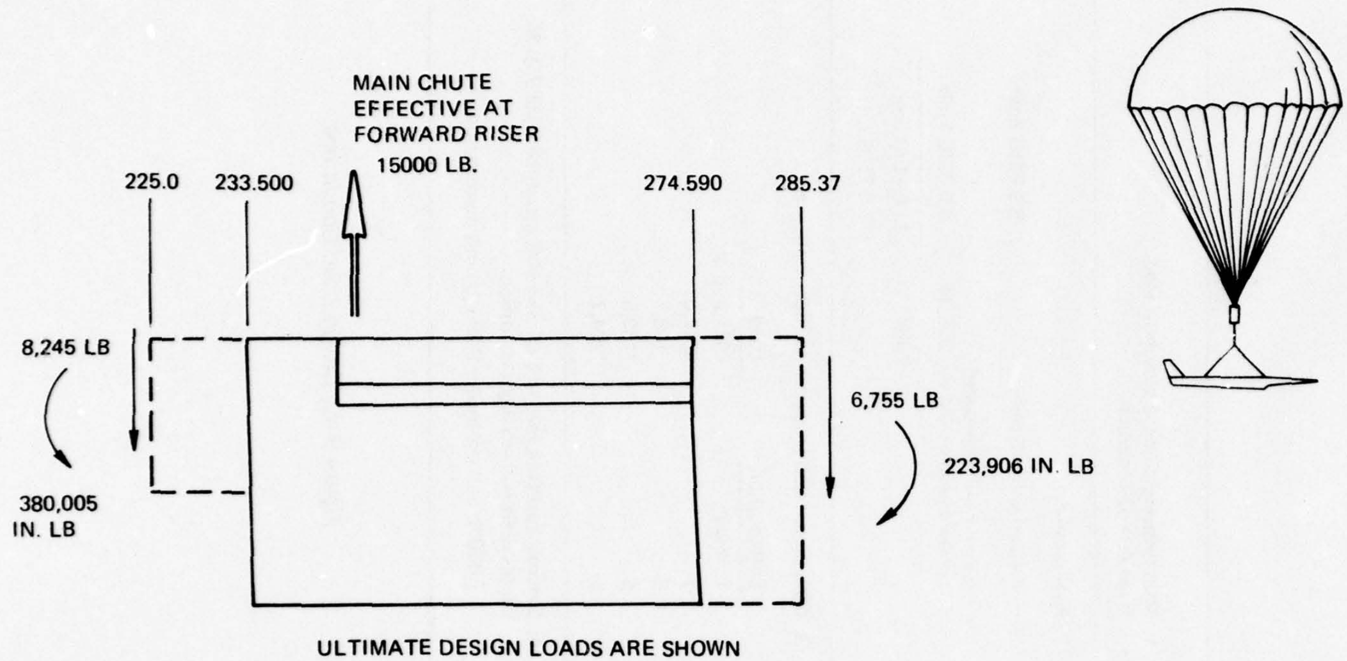


Figure 9-2. Critical Load Condition – Parachute Recovery

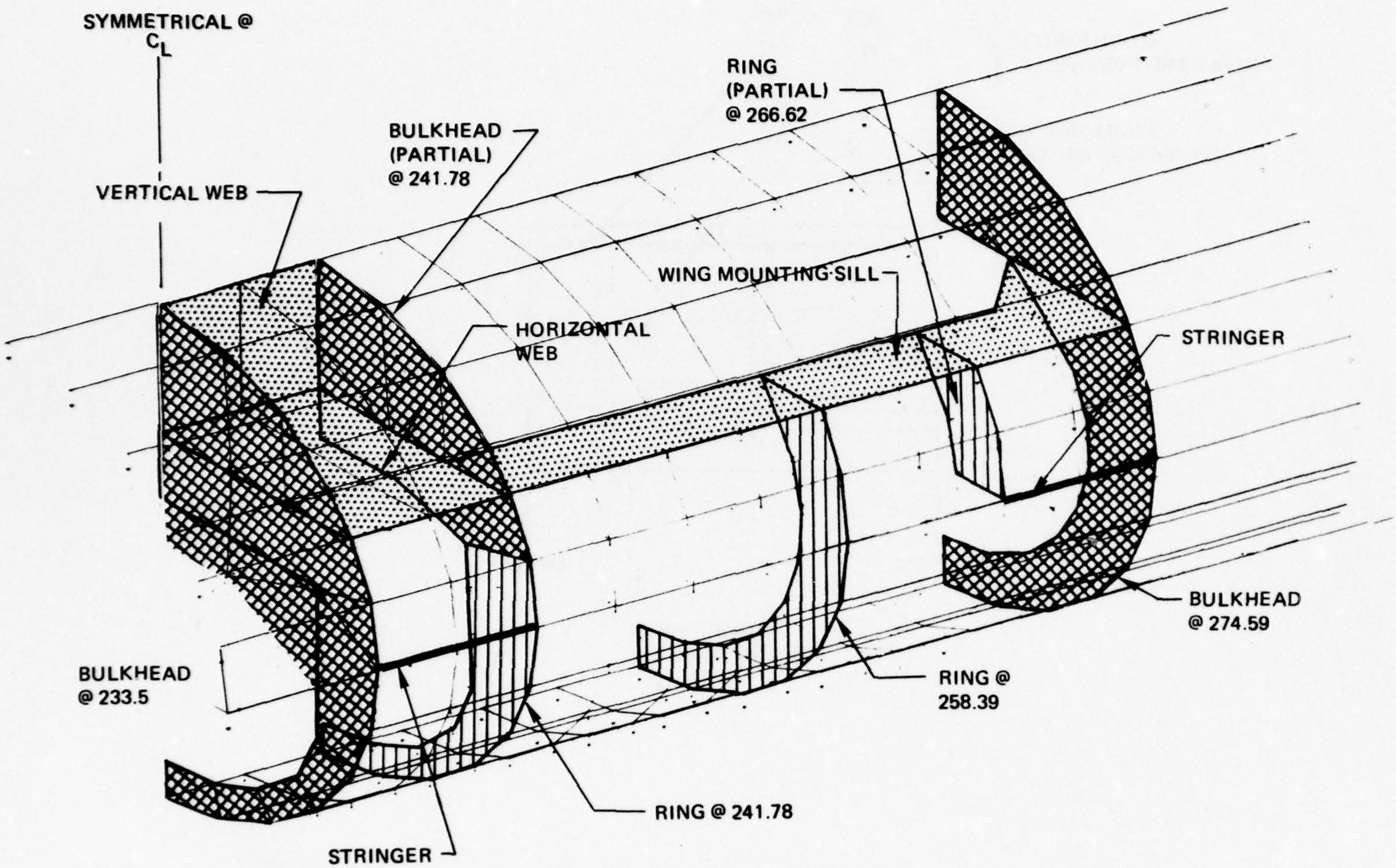


Figure 9-3. SAMECS Finite Element Model Features

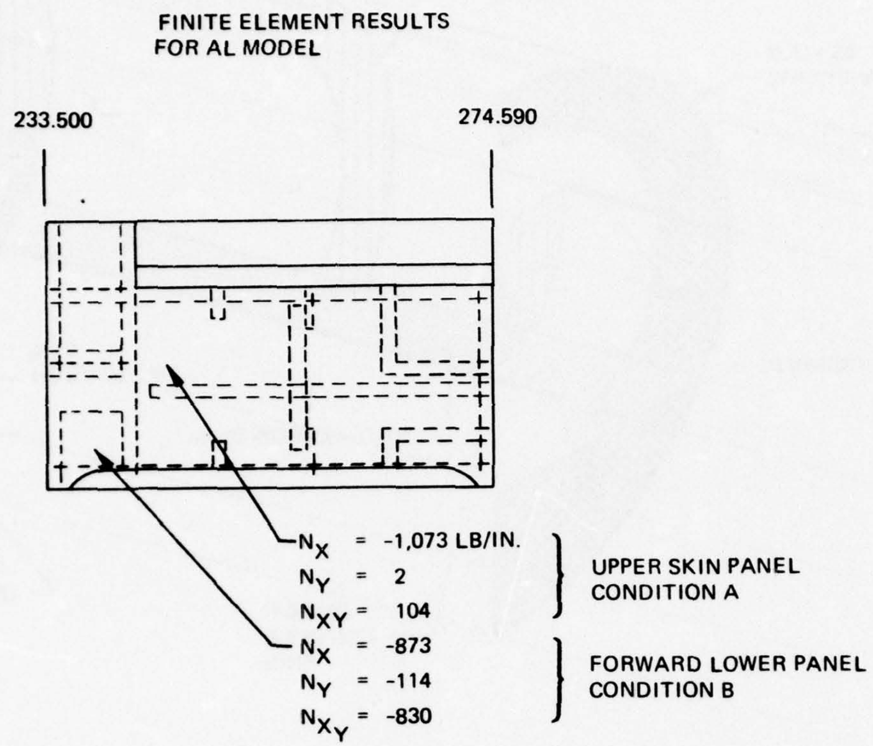


Figure 9-4. Ultimate Design Loads for Skin Laminate

9.3 PANEL BUCKLING ANALYSES

The composite centerbody section component designs are governed by local panel buckling (the subcomponent No. 2 test and STAGS-B code analysis indicated general instability will not be a governing condition). The analysis of margins-of-safety for panel buckling was accomplished using the laminate stiffness properties given in Figure 9-5, which are based on actual minimum laminate thickness, and simply supported orthotropic panel buckling analyses. Laminate properties at room temperature were computed by the classical laminate analysis method (Ref. 10). The computed theoretical panel buckling loads are shown in Figure 9-6 along with the analysis references. Three panel buckling analyses were made:

1. Upper skin panel
2. Lower skin panel
3. Lower skin panel with a full-panel doubler stock sheet

For the effective panel dimensions, the clear span dimensions between doubler edges were assumed for each panel. This appeared to be a reasonable assumption considering the high degree of skin clamping existing at the bulkhead and ring stub joints. The clamping effect, and also the tolerance to pre-buckling deformation displayed in subcomponent No. 2 test, allowed the assumption of a fairly high buckling correlation factor ("knock-down") of 0.80.

In order to have a positive margin-of-safety, the side component designs were reinforced in the lower forward side panel with a special square "doubler" patch placed at the center of the panel. The patch was the same Gr/Ps stock sheet material as the doubler sheets. The resultant critical buckling load for this partially padded panel was assumed to be the average of the cases 2a and 2b in Figure 9-6.

Undirectional ply:

$E_X = 16.E6$

$E_Y = 1.6E6$

$G_{XY} = 0.6E6$

$V_{XY} = 0.2$

$t_{ply} = 0.0048$

Stock sheet: $0, +45, -45, 90, -45, +45, 0^o$

$E_X = 6.76E6$

$E_Y = 4.97E6$

$G_{XY} = 2.69E6$

$V_{XY} = 0.42$

$t_{SS} = 0.003325$

$\epsilon_{CR\,0^o} = 9000\ \mu\epsilon$

$\epsilon_{CR\,90^o} = 9000\ \mu\epsilon$

Skin laminate: 2 stock sheets @ 0^o

$E_X = 6.76E6$

$E_Y = 4.97E6$

$G_{XY} = 2.69E7$

$V_{XY} = 0.42$

$t_S = 0.0665$

$$A_{ij} = \begin{bmatrix} 517258 & 160333 & 0 \\ & 379908 & 0 \\ S. & & 178867 \end{bmatrix} \qquad D_{ij} \begin{bmatrix} 217.02 & 54.38 & 0 \\ & 123.02 & 0 \\ S. & & 61.21 \end{bmatrix}$$

(Units are in., lb)

Figure 9-5. Laminate Properties

Panel	Thick in.	Length a, in.	Dev width b, in.	Inside radius in.	Computed critical loads lb/in.	Notes and references
(1) Upper skin panel sta 241.78 to 250.06	0.0665	5.2	7.3	12.45	$N_{XCR} = 1460.5$	NASA TND-4706 ref. 11 (typ)
					$N_{XYCR} = 1006.9$	Ref. 12 (typ)
(2) Lower skin panel (a) sta 233.5 to 241.78	0.0665	6.0	6.9	12.45	$N_{XCR} = 1384.2$ $N_{XYCR} = 876.9$	
With full pad-up (b)	0.10	6.0	6.9	12.45	$N_{XCR} = 3316.1$ $N_{XYCR} = 2379.2$	

Simple panel edges assumed

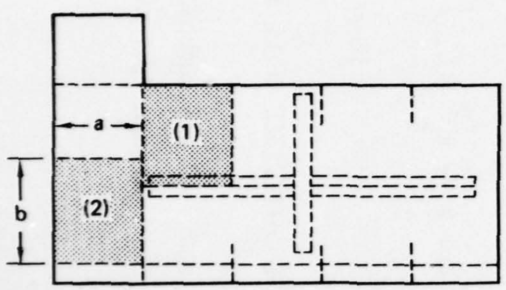


Figure 9-6. Panel Buckling Analysis Results

The classical buckling interaction relation for combined axial and shear loading was adopted. The circumferential loads (N_y) were neglected in the interaction analyses because of their low values. The resulting margin-of-safety calculations were as follows:

Upper Skin Panel:

$$N_{xcr} = 1460.5\ (0.8) = 1168.4 \text{ lb/in.}$$
$$N_{xycr} = 1006.9\ (0.8) = 805.5$$

$$R_c = 1073/1168.4$$
$$R_s = 104/805.5$$

$$R = R_c = R_s^2 = 0.935 < 1.0$$

$$MS = 1.0R - 1.0 = \underline{0.07}$$

Lower Forward Skin Panel:

$$N_{xcr} = (1384.15 + 3316.1)(0.5)(0.8) = 1880.$$
$$N_{xycr} = (876.9 + 2379.2)(0.5)(0.8) = 1302.4$$

$$R_c = 873/1180$$
$$R_s = 830/1302.4$$

$$R = R_c + R_s^2 = 0.87 < 1.0$$

$$MS = 1.0/R - 1.0 = \underline{0.15}$$

9.4 FINITE DIFFERENCE SHELL MODEL

A STAGS-B model of the composite centerbody section was developed to study the effect of combined fuel pressure and drop tank loads.

The model details (developed view) and coding are shown in Figure 9-7. The drop tank load (refer to condition 4 in Figure 9-1) was applied as an outward radial load at Row 10 and Column 1 of the model. The resulting ultimate keel beam stresses were:

σ_O = +29240 lb/in^2 in the outer flange
σ_I = -11040 lb/in^2 in the skin attachment flange

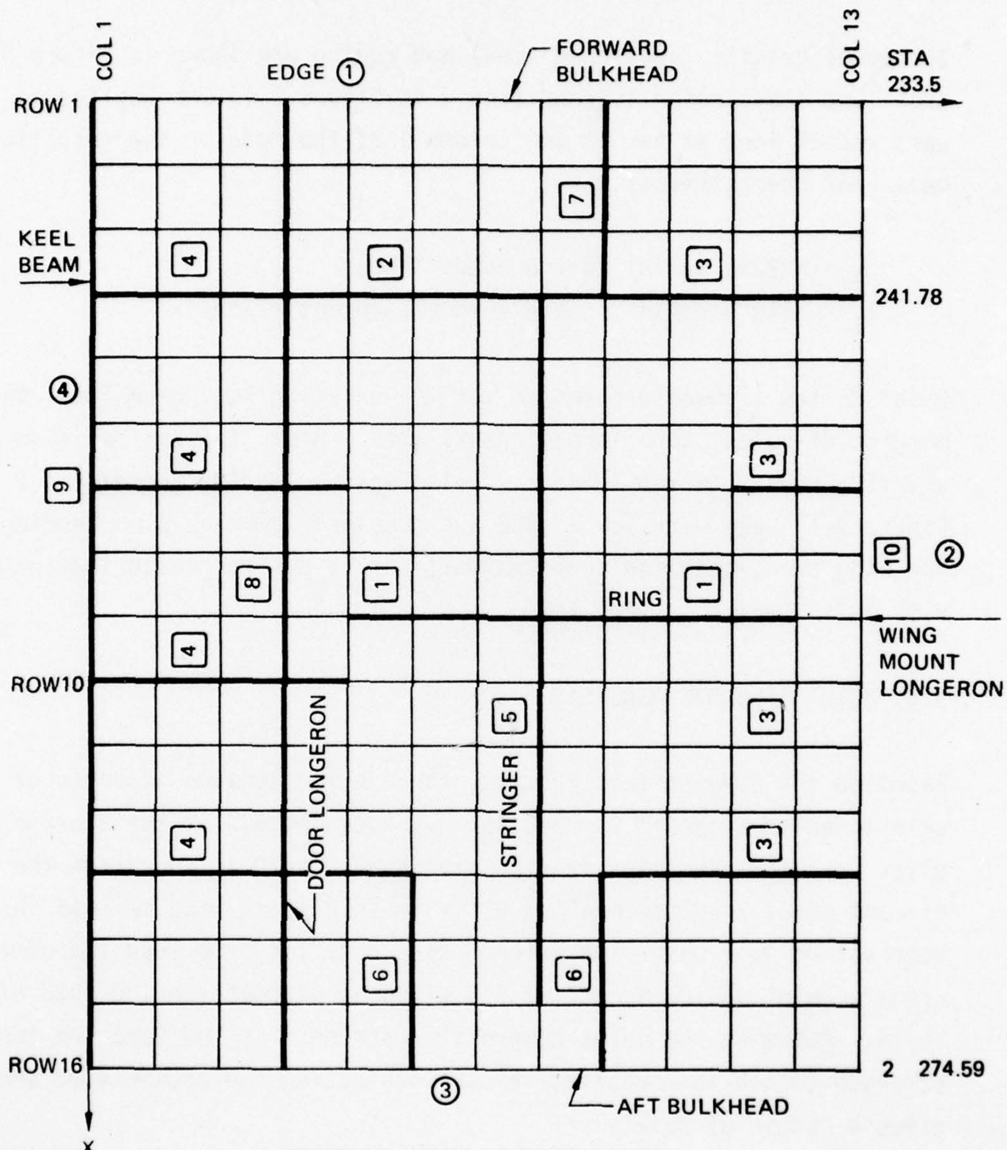
Based on the allowable stresses for the existing keel beam (Ref. 6), the margins-of-safety were large in this area. Also, the nominal skin laminate strains due to the internal fuel pressure loading (condition 2 in Figure 9-1) were very low -- 462.5$\mu\varepsilon$ combined membrane plus bending. In the ring area, the ring peel (normal) stress due to pressurization was 8.98 lb/in^2 which is very low.

9.5 JOINT STRENGTH ANALYSIS

Based on the element test results, the average tension strength of the skin-to-bulkhead joint is 3360 lb/in.; assuming 80% of the average value gives an allowable joint tension strength of 2670 lb/in. From the finite element model results, maximum ultimate skin joint load is 1449 lb/in. compression just below the forward side body longeron load introduction area; a joint design factor of 1.5 gives an ultimate design load of 2180 lb/in. Assuming the joint compression strength is the same the tension strength (which is conservative) and neglecting the presence of shear gives a margin of safety of:

MS = 2670/2180 - 1 = <u>0.23</u>

In the area of the forward side body longeron load introduction, the concentrated load at bulkhead sta. 233.50 is 6950 lb. compression as given in Ref. 6 assuming a joint design factor of 2.0 gives an ultimate design load of 13900 lb. This load is transmitted (1) to the remaining aluminum stringer section between sta. 233.50 and 241.78 and (2) to the Gr/Ps skin



EDGE	DISPLACEMENT BOUNDARY CONDITIONS			
	U	V	W	β
1	0	0	0	0
2	1	0	0	0
3	1	0	0	0
4	1	0	1	0

0 = FIXED
1 = FREE

Figure 9-7a. Stags–B Model of Centerbody Section Door and Side Panels

```
FIREBEE FILE    CENTERBODY SECTION OF BQM34E
    0    1    1
   16   13
    1   10    7    0   13    0    3
1.0       0.        1.0       1.0       0.        1.0
    1    2    0    0
42.       120.      12.65
0.        0.        0.
   16   13
    0
    0    0    0    0
    1    4    0    0
    1    0    0    0
    1    0    1    0
    1    0
10.            .    3    0    0
    1    0
500.           1    3   10    1
   10    1    3   12
    4    2    4    8
    4    3    8   13
    4    4    1    4
    7    4    1    5
   10    4    1    5
   13    4    1    5
    7    3   11   13
   10    3   13   13
   13    3    9   13
    8    5    2   15
    1    9    1   16
    4    8    1   16
    9    7    1    4
   13   10    1   16
    6    6   13   16
    9    6   18   16
    2
6640000.  0.23      0.1575    0.        0.        1184.1    0.        -0.86
0.        0.        -1.6      0.
    3
10000000. .252      0.23      0.        0.        1300.     0.        -1.32
0.        0.        -1.32     0.        -2.58     0.
    3
10000000. .347      .714      0.        0.        1788.     0.        -2.07
0.        0.        -2.07     0.        -4.08     0.
    3
10000000. .139      .031      0.        0.        975.      0.        -.82
0.        0.        -.82      0.        -1.58     0.
    3
10000000. .2        .0565     0.        0.        700.      0.        -.68
0.        0.        -.68      0.        -1.58     0.
    3
10000000. 0.168     0.0136    0.        0.        1398.     0.        -0.41
0.        0.        -0.41     0.        -1.18     0.
    3
10000000. 0.319     0.0435    0.        0.        6475.     0.        -0.362
0.        0.        -0.362    0.        -1.38     0.
    3
10000000. 0.237     0.0013    0.1645    0.        2000.     0.        -0.187
0.        0.        -0.137    0.        -0.374    0.
    3
100000000. 0.326    0.211     0.        0.        0.        0.        +0.545
0.        0.        +0.545    0.        +.13      0.
    3
10000000. 0.234     0.08      0.08      0.        272.      0.        -0.5
0.        0.        -0.5      0.        -2.0      0.
    1    1    1
    1    0    0
UNIFORM ORTHOTROPIC WALL
    1    0
    1
564593.   168323.   414217.   186499.
346.4     91.       203.5     202.9
42.       120.
```

Figure 9-7b. STAGS-B Model of Centerbody Section Door and Side Panels

laminate. Subcomponent No. 1 was loaded to 44682 lb. without failure in the joint area but this subcomponent had heavy reinforcement in the load introduction area. Subcomponent No. 2 did not have the extra metallic joint reinforcement and was tested with a concentrated load to 6899 lb. without failure which was essentially a test to ultimate load. Based on these subcomponent tests, the concentrated load capability of the composite components is conservatively estimated to be 20000 lb. The resulting margin-of-safety calculation for the concentrated loading is:

$$MS = 20000/13900 - 1 = \underline{0.44}$$

9.6 MARGIN-OF-SAFETY SUMMARY

A summary of the calculated margins-of-safety for the delivered components is given in Figure 9-8. The most critical area is buckling of the upper skin panel between sta. 241.78 and 250.06 which has a margin-of-safety of +0.07 relative to the ultimate design load condition 4PX02. The door component is not shown in the summary because large margins-of-safety exist in all areas due to low internal loads.

9.7 WEIGHT ANALYSIS

A weight analysis summary is presented in Figure 9-9. The final actual weight savings was only 5% due to conservatism in the door component design, a heavy ring stiffener concept, and use of heavy doublers required by fastener constraints. For comparison purposes, the weight summary includes a hypothetical design configuration which lacks doublers and ring honeycomb sealant; the associated weight saving is 37%. In a new design, without the XBQM-34E retrofit constraints encountered in this program, a weight savings of 15 to 20% could be reasonably expected.

	Area	Margin of safety	Basis
1.	Upper skin panel buckling Station 241.78 to 250.06	0.07	Panel buckling analysis
2.	Lower skin panel buckling Station 233.5 to 241.78	0.15	
3.	General skin instability	Large	Subcomponent No. 2 test
4.	Panel strain due to fuel pressure	4.66	Stags-B model results for fuel pressure and drop tank loads
5.	Ring peeling from skin due to radial stress	Large	
6.	Skin-to-bulkhead joint at 233.5	0.23	Joint element tests and finite element model results
7.	Forward longeron load introduction details	0.44	Subcomponent No. 1 and 2 tests
8.	Wing mounting bolt load introduction details	Large	Joint element tests and finite element model results
9.	Keel beam outstanding flanges in tension	1.63	Stags-B model results for fuel pressure and drop tank loads and Ryan structural analysis report TRA 16642-13B
10.	Keel beam flanges in compression	3.2	

Figure 9-8. Margins-of-Safety Summary

Item	As-Built	No Doublers Or Potting Included
New Gr/Ps components		
Left skin	5.49 lb	3.72
Right skin	5.63	4.10
Door	2.57	2.02
Total	13.69	9.84
Removed aluminum parts	-14.40	-14.40
Ballast correction	0	0
Net weight change	-0.71	-4.56
Weight saving	5%	32%

Note: Fastener weights not included
As-built Gr/Ps component weights are actuals

Figure 9-9: Gr/Ps Component Weight Comparisons

10.0 COMPONENT TEST PROGRAM

Two test conditions were presented in a separate test plan (Ref. 13) which identified requirements for the ground testing of the delivered components on a surplus XBQM-34E by NAVAIRDEVCEN. The conditions were:

1. Main recovery chute deployment limit load test
2. 5g maneuver limit load test

These two conditions were selected for initial certification of the design concept and is discussed in this section. The other test conditions would be required in case of eventual flight test (Ref. 13).

An entire XBQM-34E fuselage was tested in a self-contained test fixture fabricated from 12 in. wide flange beams. Loading was accomplished at selected fuselage frame stations by means of continuous aluminum bands supported on 3/4" thick elastomer compression pads. Where obstacles such as the vertical stabilizer prevented use of continuous bands, aluminu straps were bonded to the fuselage with RTV-88 silicone rubber. Separate wiffle tree arrangements were used to test each of the two load conditions and were attached to the fuselage frame loading points. In both tests, loads were introduced into the structure through a single load point by means of a manually operated hydraulic jack. Figures 10-1 and 10-2 are photographs of the test set-up.

Figure 10-3 defines the procedure that was followed by NAVAIRDEVCEN. Figures 10-4 and 10-5 identify the test load locations and their respective magnitudes. The resultant airframe moment distribution is given in Figure 10-6 in addition to loads developed in Teledyne Ryan tests during the BQM-34E development program (Ref. 14). The recovery test condition closely simulated the corresponding actual main chute deployment static test condition (Teledyne Ryan test 5 in Figure 10-6) and the corresponding 4PX02 design load condition that was treated in the structural analysis (discussed in the preceeding section).

Strain data was recorded during the NAVAIRDEVCEN testing using strain gages (located as illustrated in Figure B-1 of Appendix B). The gages were installed on the delivered components by Boeing at the center of selected panels where pre-buckling strains can occur and at the edges of panels where strain concentrations may develop. Also, gages were placed on the stiffeners to monitor bending deformations. Post-test analysis of the recorded strain data is reported in Appendix B.

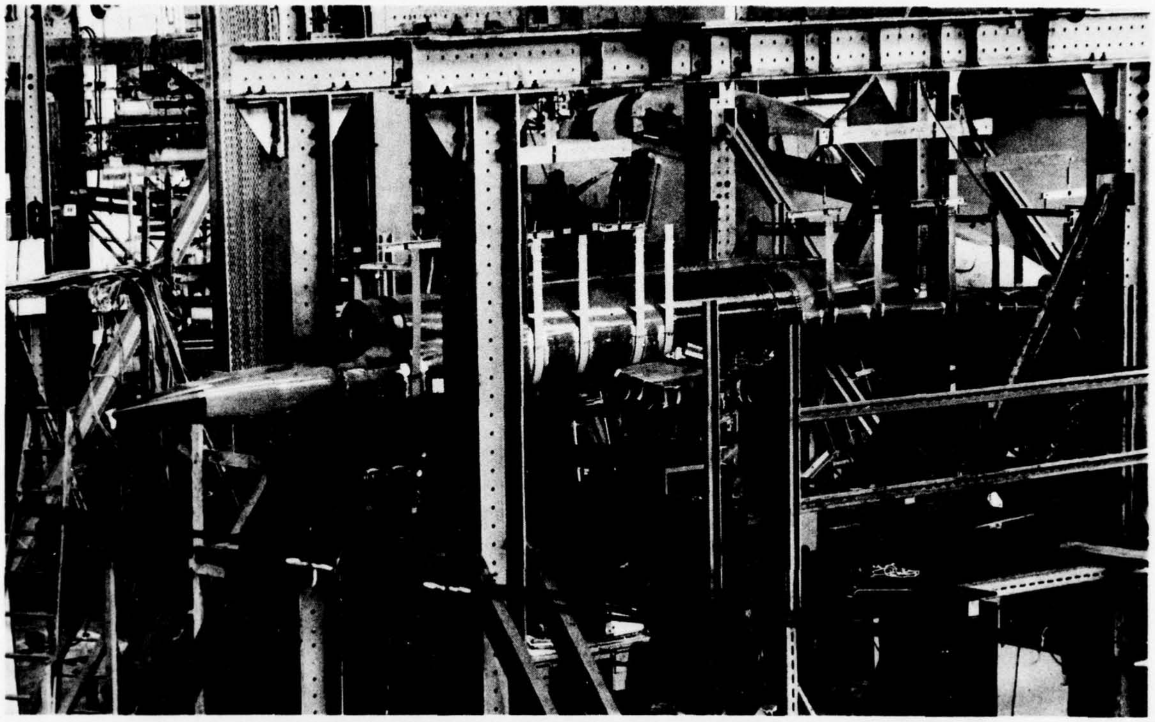


Figure 10-1: XBQM-34E in Test Fixture

Figure 10-2: XBQM-34E Wing Loading System

Procedure:

A. Apply 25% limit load

- Checkout instrumentation and test set-up

B. Apply 50% limit load 10 times

- Extrapolate strain data to limit load and compare first and last cycle data

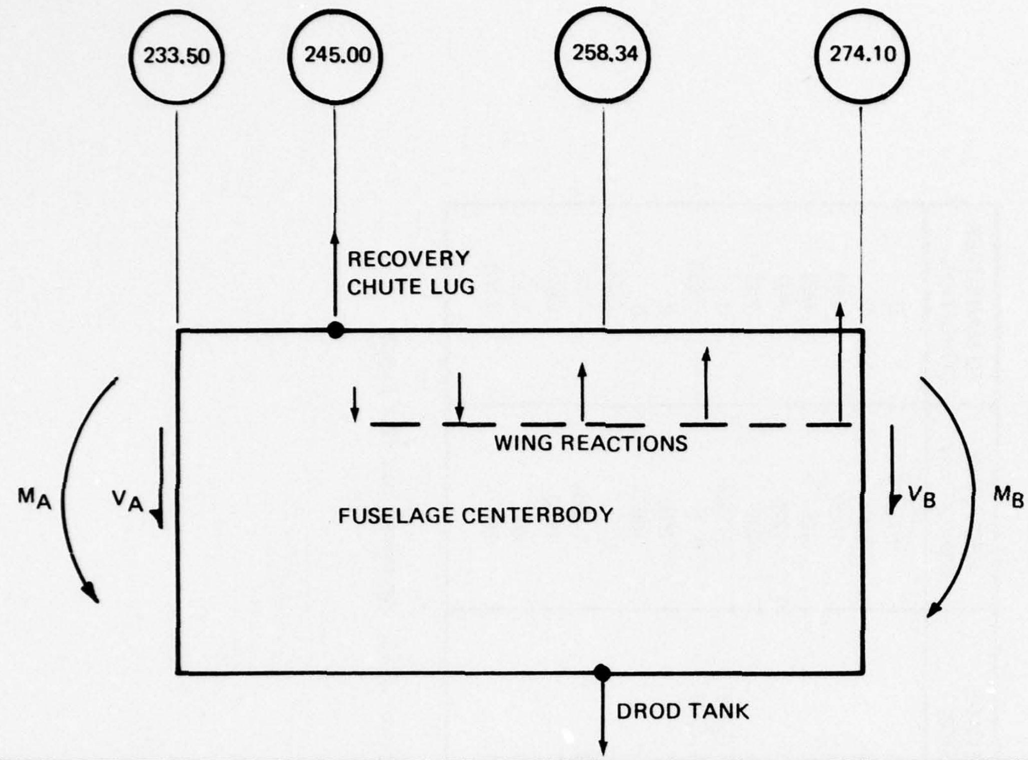
C. Apply limit load

- Record strain data
- Evaluate strains during loading using force/strain procedure

D. Apply limit load 5 times

- Record data

Figure 10-3: Test Loading Procedure



CONDITION	M_A IN-LB	M_B	V_A LB	V_B
RECOVERY	292940	166040	5117	6383
5g MANEUVER	90780	227000	2000	3860

Figure 10-4: Fuselage Centerbody Limit Body Loads

FUSELAGE STATION	RECOVERY CONDITION	5G MANEUVER CONDITION
118.50	-575 LB.	0
134.30	-500	0
166.30	-1500	-650
182.50	-825	-650
209.00	-1228	-497
224.80	-489	-203
245.00 (LUG)	+11500	0
258.34 (TANK)	0	-2950
274.10	-1487	0
285.20	-1096	0
WING	0	+8810
301.90	-1400	-798
315.20	-1500	-362
325.00	-600	-600
350.00	-300	-2100

Figure 10-5: Applied Limit Test Loads

AD-A035 398 BOEING AEROSPACE CO SEATTLE WASH F/G 1/3
DEVELOPMENT OF A LOW-COST GRAPHITE REINFORCED COMPOSITE PRIMARY--ETC(U)
DEC 76 J H LAAKSO, J T HOGGATT N622669-74-C-0368

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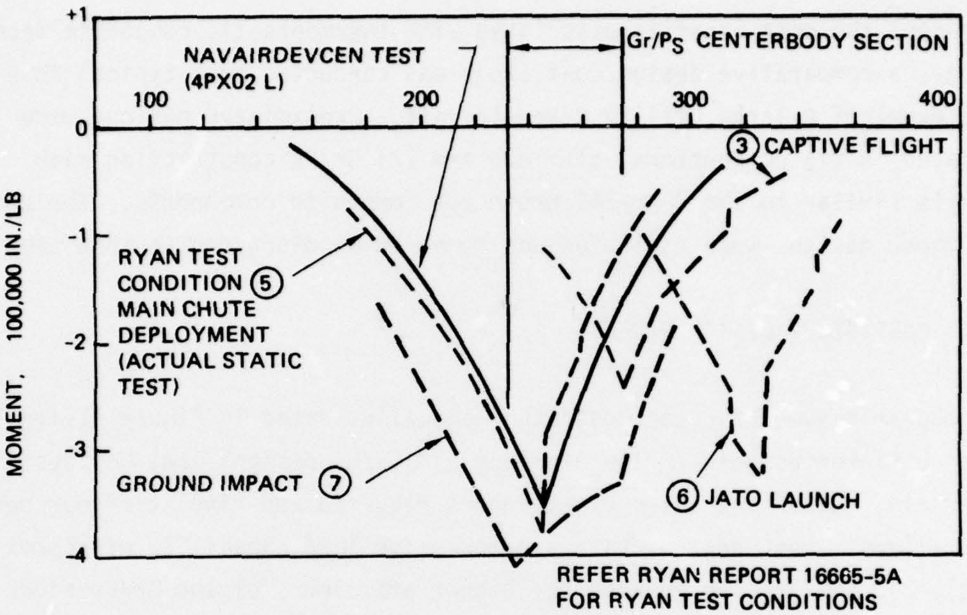


Figure 10-6: XBQM–34E Test Condition Moments

11.0 COMPARATIVE COST STUDIES

To assess the cost benefits associated with thermoplastic composite technology, a comparative design/cost study was conducted on a typical fuselage panel of a large utility type aircraft. Preliminary designs were prepared in (1) conventional aluminum and (2) Gr/Ps construction with details similar to the XBQM-34E prototype composite components. The costs for these designs were estimated and compared as discussed in this section.

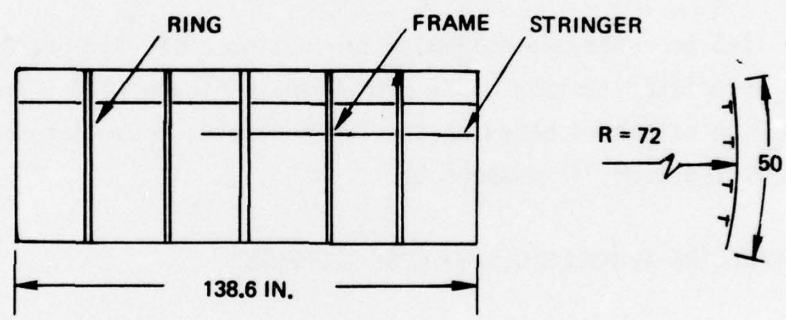
11.1 COST STUDY DESIGN MODELS

The models assumed for the cost study are illustrated in Figure 11-1. The basic differences between the aluminum and Gr/Ps designs are, besides materials, number and types of stringers required and ring stiffener configurations. Both designs have a compressive load capability of approximately 3400 lb/in. Because of the higher efficiency of the Gr/Ps stock sheet skin concept, the stringer spacing for this design is increased which reduces stringer part count. The Gr/Ps design has Gr/Ps ring stiffeners and also has two aluminum ring frames to provide conventional connections for other interior airframe hardware.

Preliminary design drawings are given in Appendix A for the respective designs (Dwgs. SCRR21475 for aluminum and SCRR22075 for Gr/Ps). Two composite ring concept alternatives are shown: a honeycomb concept and an I-section concept like the one discussed in the Fabrication Processes Section (Section 5.5). The I-section ring was selected for the following cost studies and is judged to be a practical concept from a durability point-of-view. The composite stringers and skin are essentially the same as the delivered Gr/Ps components. The aluminum design details are typical of large commercial aircraft. Based on the design drawings, the weights of the designs are compared in Figure 11-1.

Design model:

Representative unpressurized body panel structure for large transport aircraft



Design	No. Al frames	No. composite rings	No. stringers	Wt lb
Al	5	0	7	100
G/PS	2	3	5	83.8

Figure 11-1: Manufacturing Cost Comparison Case Study

11.2 Gr/Ps DESIGN COST ELEMENTS

A fabrication sequence for the Gr/Ps fuselage panel design was defined based on the extrapolation of the XBQM-34E component fabrication processes and some new but potentially practical ideas. These fabrication process steps are shown in Figures 11-2 to 11-4. Additional manufacturing development (not a great deal) is needed to establish the feasibility of the ring joggling and the reusable silicone rubber pressure pad method of fusing the final panel assembly together (versus the epoxy bonding method used for the delivered components).

Figure 11-5 presents the estimated production costs for one Gr/Ps panel unit. Associated tooling costs are shown in Figure 11-6. These cost are engineering estimates based on the labor records accumulated during fabrication of the XBQM-34E components.

11.3 BASELINE ALUMINUM DESIGN COST ELEMENTS

The production costs associated with the aluminum fuselage panel design were estimated by experienced cost estimators, using standard procedures, and are itemized in Figure 11-7. The production costs shown are for one unit.

11.4 PRODUCTION RUN COST ANALYSIS

A production run cost analysis was made for the metal and composite fuselage panel designs based on the assumptions listed in Figure 11-8. Tooling fabrication costs for the composite design were close to standard factored costs for the aluminum design 3004 vs. 3547 basic tooling fabrication hours) so the respective costs were assumed to be the same (3547 BTF hours). Standard factors were assumed for the factored labor items indicated in the Figure 11-8.

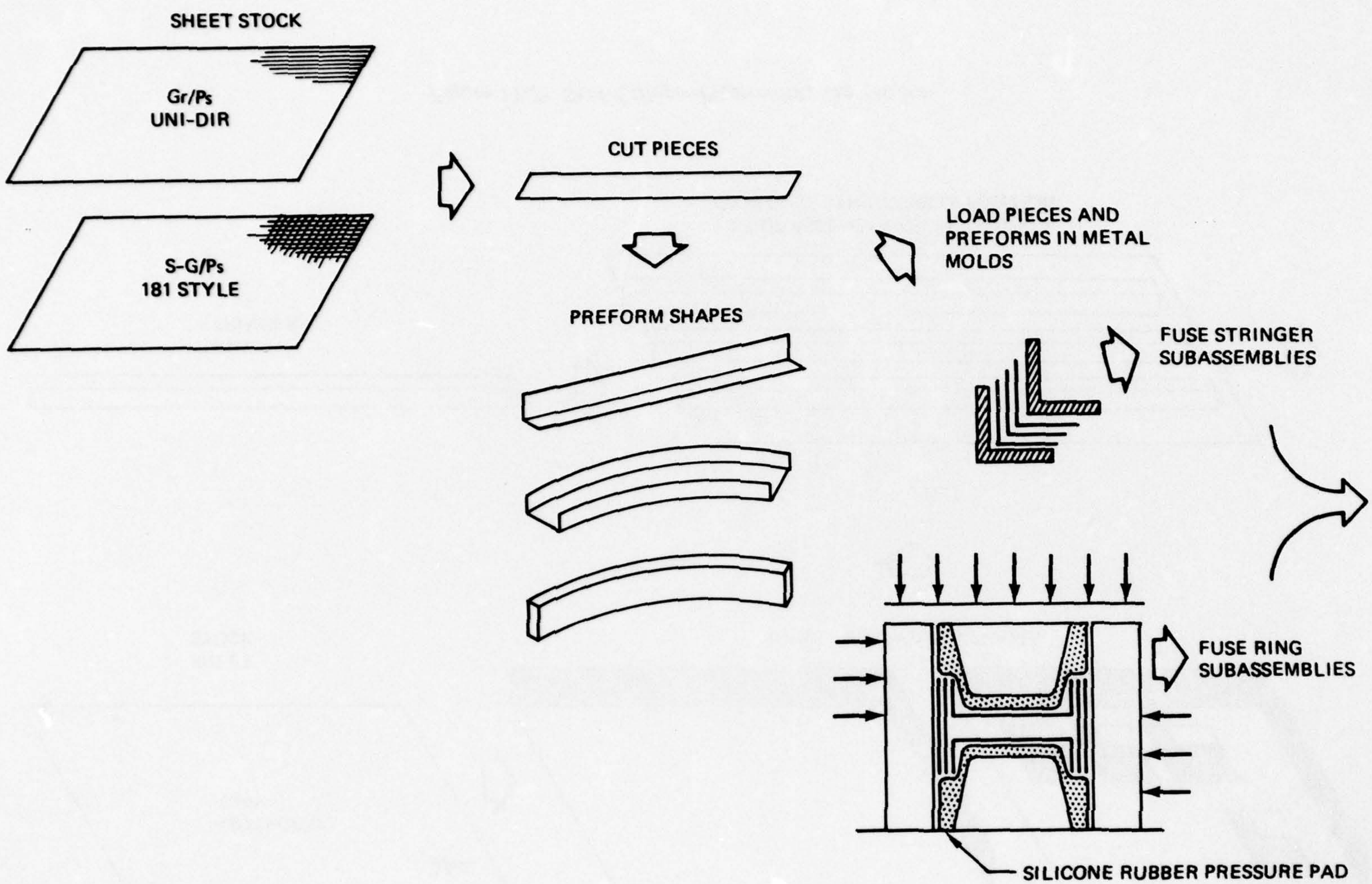


Figure 11-2 Stiffener Subassembly Operations

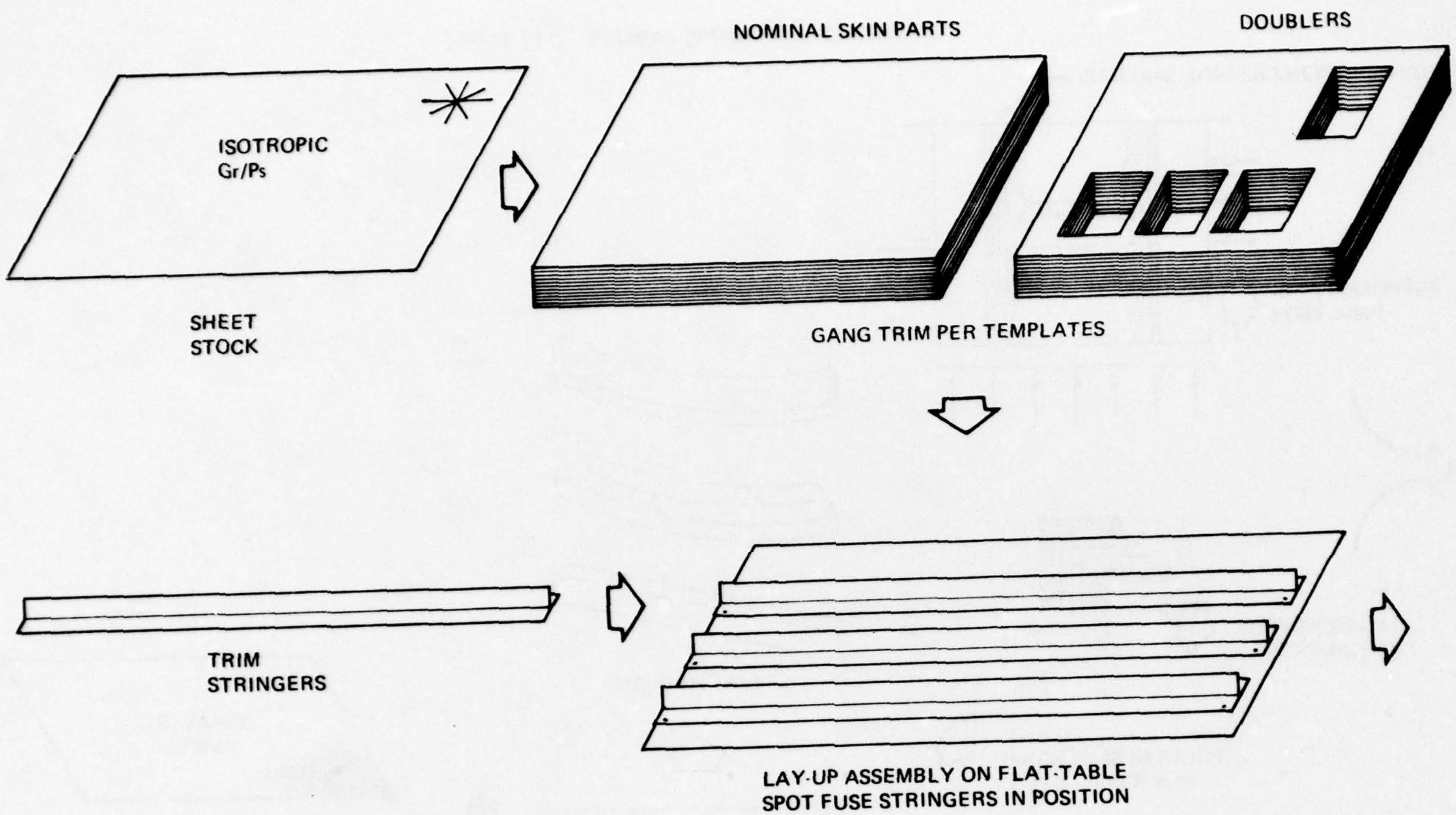


Figure 11-3: Skin-Stringer Preassembly Operations

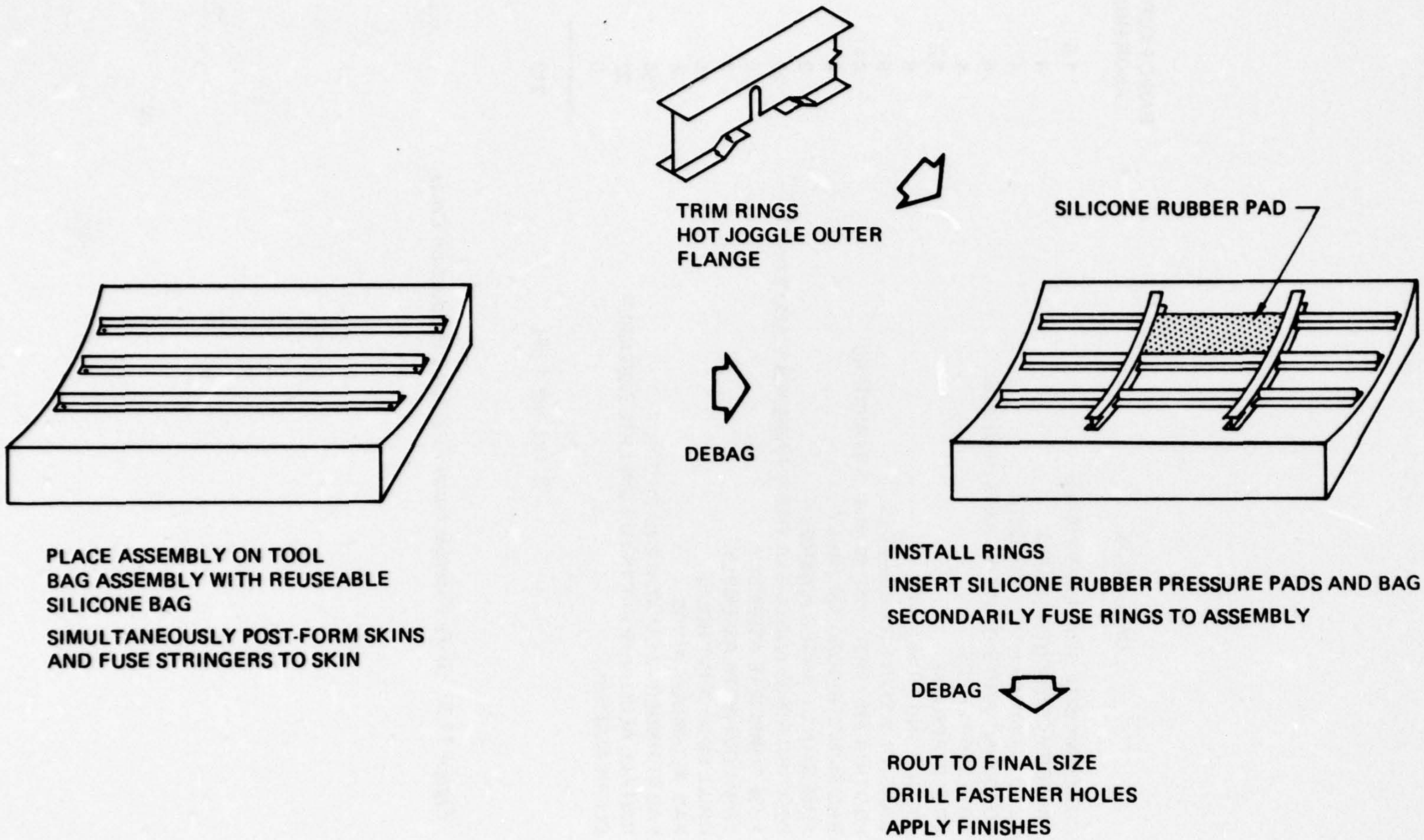


Figure 11-4: Final Assembly Operations

	OPERATION	BASIC FACTORY LABOR HOURS
1.	CUT COMPOSITE STIFFENER PIECES	1.5
2.	PREFORM STIFFENER SHAPES	4
3.	LOAD STRINGER PARTS IN MOLDS	4
4.	LOAD RING AND SHEAR TIE PARTS IN MOLDS	4
5.	FUSE STIFFENERS	4
6.	TRIM STRINGERS	1.5
7.	JOGGLE AND TRIM RINGS	6
8.	GANG TRIM SKIN STOCK SHEETS	8
9.	POSITION AND SPOT-FUSE SKIN AND STRINGERS	4
10.	BAG SKIN/STRINGER ASSEMBLY	6
11.	FUSE SKIN/STRINGER ASSEMBLY	8
12.	POSITION RINGS AND SHEAR TIES ON ASSEMBLY AND BAG	20
13.	FUSE COMPOSITE ASSEMBLY	4
14.	TRIM COMPOSITE ASSEMBLY	3
15.	DRILL EDGE JOINT HOLES	15
16.	FAB. ALUMINUM RINGS	4
17.	FAB STRINGER ALUMINUM END FITTINGS	154
18.	INSTALL ALUMINUM PARTS TO COMPLETE ASSEMBLY	22
19.	CLEAN ASSEMBLY	6
	TOTAL NO. 1 BFL	279

Figure 11-5: Gr/Ps Fuselage Panel First Unit Production Costs

ITEM	BASIC FACTORY LABOR HOURS
1. Stiffener Preform Tooling	100
2. Multi-Cavity Molds for Fusing Ring and Shear Tie Parts	400
3. Molds for Stringers	120
4. Skin Part Cutting Templates	40
5. Trimming and Positioning Templates for Stiffeners	100
6. Ring Joggling Dies	120
7. Cutter Set-Ups	24
8. Skin Lay-Up Plate Tool	40
9. Intermediate Bag Fabrication	200
10. Assembly Fusing Tooling	1400
11. Assembly Fusing Bag	360
12. Router and Drill Templates for Final Assembly	100
TOTAL	3004

Figure 11-6. Gr/Ps Fuselage Panel Tooling Fabrication Costs

	OPERATION	BASIC FACTORY LABOR HOURS
1.	Shear, Stretch Form and Rout Skins	6
2.	Yoder Roll, Buffalo Roll and Trim Rings	10
3.	Yoder Roll and Trim Stringers	6
4.	Roll Form and Saw Shear Ties	12
5.	Break Form and Saw Clips	8
6.	Trim Straps	2
7.	Blank and Form Gussets	12
8.	Machine Joint Fitting Forgings	216
	TOTAL DETAIL FAB.	272
9.	Assembly and Clean	156
	TOTAL NO. 1 BFL	428

Figure 11-7. Aluminum Fuselage Panel First Unit Production Costs

- 85% learning curve for production labor
- Tooling costs are based on standard factors for the aluminum design and are assumed to be the same for the composite design
- All composite materials are furnished by vendors in stock sheet laminate form
- Material allowances are included in the assumed material costs
- Q/C and tooling material allowances are included in the assumed labor rate for fabrication and tooling
- Factored labor items:

    Fabrication pickups
    Fabrication rework
    Tooling & production planning
    Distributed direct
    Production control and records (PCR)
    Tool design
    Tool tryout and rework
    Tool maintenance

Figure 11-8: Cost Analysis Assumptions

Two key assumptions were made in the cost analysis:

1. Gr/Ps stock sheet was purchased at $25.00/lb.

2. Labor rate for production and tooling was $50.00 per burdened basic factory labor (BFL) hour.

The results of the production run cost analysis are presented in Figures 11-9 and 11-10. The first unit total costs of each design are essentially equal (a coincidence) because the Gr/Ps design has lower fabrication hours but higher material cost than the aluminum design. As the learning curve (85%) becomes influential with increasing production units, the Gr/Ps design offers costs savings of varying amounts (26% at 100 units). However, the Gr/Ps design has higher costs after about 700 units because of the (1) basic difference in material price and (2) diminishing labor cost fraction (due to the learning curve assumption).

While the assumptions used in this cost analysis are subject to debate, the results of the study indicate:

1. Gr/Ps components have a potential for offering cost saving, as well as weight saving, depending on the design application and process development status.

2. The amount of cost saving will depend on the actual labor rates, material purchase price, fabrication hours and design configuration.

In the future (that all composite cost studies refer to), an increase in the labor rate will increase the cost savings and delay the cost cross-over point shown in this study, providing the material price is achieved. It is of importance to future application studies that there will be a cost cross-over point as long as a significant difference exists in the

basic material price of graphite reinforcements and aluminum. A parallel situation exists in the automotive industry wherein automobile bodies produced in high volume are made from low cost steel and not glass-reinforced plastic.

	NO. OF UNITS	TOTAL PRODUCTION		TOTAL TOOLING FAB		MATERIAL			TOTAL ($)	RELATIVE COST
		HRS	($) @ 50.00	HRS	($) @ 50.00	LB	$/LB	($)		
AL DESIGN	1	2,837	141,850	4,583	229,150	100 AL	2.00	200	371,200	1.0
	10	6,467	323,350	4,677	233,850	1,000	2.00	2,000	559,200	1.0
	100	29,025	1,451,250	5,185	259,250	10,000	2.00	20,000	1,730,500	1.0
	1,000	143,574	7,178,700	7,820	391,000	100,000	2.00	200,000	7,769,700	1.0
COMPOSITE DESIGN	1	2,261	113,050	3,880	194,000	6.7 AL 77.1 COMP	2.00 25.00	13 1,427	308,990	0.83
	10	4,654	232,700	3,948	197,400	67 AL 771 COMP	2.00 25.00	134 19,275	449,509	0.80
	100	19,367	968,350	4,350	217,500	670 AL 7,710 COMP	2.00 25.00	1,340 192,750	1,379,940	0.80
	1,000	94,214	4,710,700	6,451	322,550	6,700 AL 77,100 COMP	2.00 25.00	13,400 1,927,500	6,974,150	1.25

Figure 11-9: Large Scale Fuselage Component Cost Analysis Data

12.0 DESIGN CONCEPT EVALUATION CONCLUSIONS

The graphite reinforced thermoplastic design concept, so far as developed in this program, has shown to have promise for light, low-cost primary aircraft fuselage structure. The sheet stock concept for Gr/Ps skin laminates is interesting from the standpoints of a convenient vendor material form, design versatility, low-cost production thermoforming, inspectability, and reparability. The stringer sections that were developed are also attractive, because of their simplicity, for production hardware. The honeycomb sandwich ring stiffener concept used on the prototype components, however, is not suitable for general applications because of vulnerability to damage; with additional development, the other thermoplastic ring concepts that were briefly studied would be preferred candidates.

From a weapon system performance point-of-view, the Gr/Ps design concept offers potential benefits similar to epoxy-based composites: weight savings, fatigue resistance and corrosion resistance. Of interest with respect to durability is the toughness exhibited by the Gr/Ps specimens and components fabricated in this program. An area of concern is the susceptibility of polysulfone to attack by certain solvents (Ref. 1); this problem could be solved by fusing a cladding of solvent resistant resin on all exposed surfaces (in an IRAD program, Boeing successfully fused a Gr/Ps laminate with polyphenylene sulfide cladding).

Additional development of fabrication processes is required before the cost saving potential of thermoplastics predicted in the comparative cost studies can be realized. Final assembly of skins and stiffeners by integral fusing (such as by the silicone rubber pressure pad approach shown in Figure 11-4) would result in a significant cost advantage for the thermoplastic composites. Tooling development is needed to assume uniform pressure hardware quality. In addition, the ring concept and cladding development suggested above is necessary for design practicality.

The development activities in this program were restricted to cylindrical fuselage sections because of funding limitations. Obviously, the potential merits of thermoplastic composites need to be demonstrated in parts having double curvature which is typical of aircraft contours. In order to achieve post-formability, modifications to the stock sheet configurations developed in this program will be required.

13.0 REFERENCES

1. J. T. Hoggatt, "Investigation of Reinforced Thermoplastics for Naval Aircraft Structural Applications," D180-17531-1, Boeing Aerospace Company, May 1973, Prepared under NASC Contract N00019-72-C-0526.

2. J. T. Hoggatt, "Study of Graphite Fiber Reinforced Thermoplastic Composites," D180-18034-1, Boeing Aerospace Company, February 1974, Prepared under NASC Contract N00019-73-C-0414.

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4. 3-M Bonding Film AF-143 (XA-143) Introductory Data Sheet. August 4, 1972.

5. M. S. Rosenfeld, "The Effect of a Corrosive Environment on the Strength and Life of Graphite-Epoxy Mechanically Fastened Joints," Presented at the Program Review of Navy Sponsored Work on Composite Materials, Naval Research Laboratory, Washington D.C., March 4-6, 1975.

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7. B. O. Almroth, et al, "User's Manual for the STAGS Computer Code," Lockheed Missiles and Space Company Report LMSCD266611 (STAGS-A), October 1972.

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9. Correspondence from NAVAIRDEVCEN, M. S. Rosenfeld, September 23, 1974, discussing wing mounting bolt loads.

10. NASA CR-224, "Strength Characteristics of Composites," S. W. Tsai, April 1965.

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14. Teledyne Ryan Aeronautical Report No. 16665-5A, Static Test Results for XBQM-34E Supersonic Aerial Target (including Addendum I), C. Conaway, Prepared under Naval Air Systems Command Contract No. 65-0652-ci, March 1, 1968.

APPENDIX A

Design Drawing	Assembly
SK-0312750KO	Skin Installation - Centerbody Section
SK30675JL	Details -- Frame Section -- Centerbody (XBQM-34E)
SK21775KO	Forward Longeron Load Distribution Panel -- Subcomponent No. 1
SK22175KO	Subcomponent No. 2
SSRR21475	Typical Aluminum Alloy Body Structure (Unpressurized)
SCRR22075	Typical Graphite Reinforced Polysulfone Body Structure (Unpressurized)

6 | 5 | 4

D

7 LAMINATE SHALL BE 9 PLIES OF SGF/PS (181 CLOTH) W
LAYUP ORIENTATION.

8 LAMINATE SHALL BE 15 PLIES UNIDIRECTIONAL
PER 4

9 AF-143 EPOXY FILM ADHESIVE PER BMS 5-104. WT. .1
FAYING SURFACES. SCOTHBRITE/ALCOHOL CLEAN. C
325-355°F @ WITH VACUUM BAG.

10 AF-143 EPOXY FILM ADHESIVE PER BMS 5-104. WT
FAYING SURFACES. SCOTCHBRITE/ALCOHOL CLEAN. CU
325-355°F @ WITH VACUUM BAG.

11 ZINC CHROMATE PRIME

12 MILL -12 TO FIT STEP UP WITH -4 ASSY

13 LAMINATE ASSEMBLIES PER FABRICATION PLAN CS 2

14 APPLY EPOXY FOAMING ADHESIVE (BMS 5-90
AT RING (-11 ASSY) STRINGER (-4 ASSY) INTERSECTION
.25 MIN FILLETS

15 SEAL ALL H/C CORE SIDE EDGES & ENDS WITH
TYPE 5 EPOXY ADHESIVE

16 -10 INSTALLATION PER RYAN DRAWING 166FO
INSTL - FUEL TANK FUSELAGE) - 9 ASSY WILL REP
166F098-21 (DOOR).

C

B

A

PART NUMBER	NEXT ASSY	USED ON	EFFECTIVITY	DWG BY NUMBER	REV LTR
		APPLICATION			

6 | 5 | 4

3 | 2 | 1

L BE 9 PLIES OF SGF/PS (181 CLOTH) WITH 45°
ION.

L BE 15 PLIES UNIDIRECTIONAL G/PS

LM ADHESIVE PER BMS 5-104. WT • .10 PSF (.014 MIL)
S. SCOTHBRITE/ ALCOHOL CLEAN. CURE 1HR @
ITH VACUUM BAG
LM ADHESIVE PER BMS 5-104. WT = .05 PSF (.007 MIL)
S. SCOTCHBRITE/ALCOHOL CLEAN. CURE 1HR @
ITH VACUUM BAG.

PRIME

STEP UP WITH -4 ASSY

BLIES PER FABRICATION PLAN CS 25540-FP-1

FOAMING ADHESIVE (BMS 5-90)
STRINGER (-4 ASSY) INTERSECTION WITH
S

HC CORE SIDE EDGES & ENDS WITH BMS 5-28
OXY ADHESIVE

TION PER RYAN DRAWING 166F098 (DOOR
NK FUSELAGE) -19 ASSY WILL REPLACE
OOR).

NOTES

1. SPECIFICATIONS AND STANDARDS FOR THE CONTROL OF MANUFACTURING OPERATIONS (AS APPLICABLE)
FORM, STRAIGHTEN & FIT METAL PARTS PER BAC [illegible] — BOLT & NUT INSTALLATION PER BAC [illegible]
RIVET INSTALLATION & SYMBOLS PER BAC [illegible] — PART MARKING PER BAC [illegible]
FINISH CODES PER DOCUMENT [illegible]

2. INSTALL -2 & -3 SKIN ASSEMBLIES USING 3/16 DIA, 100° C'SUNK HD, STEEL BLIND BOLTS, HUCK TYPE RECOMMENDED, USING THE EXISTING HOLE LOCATIONS ON THE MODIFIED STRUCTURE.

3. NOMEX 3/16 CELL HONEYCOMB PER BMS 8-124, CLASS IV, TYPE I, HEAT FORMED PER BAC 5201

4. G/PS TYPE II PRELIMINARY MATERIAL SPECIFICATION, UNIDIRECTIONAL PREIMPREGNATED TAPE, THERMOPLASTIC MATRIX, JUNE 1974 PS MATERIAL IS UNION CARBIDE P1700.

5. LAMINATE SHALL BE 1 STOCK 7 PLY SHEETS (0°:45°:90°:45°:0°) OF G/PS PER 4, 1 FILLER PLY OF SGF/PS (181) CLOTH. SEE DETAIL

6. LAMINATE SHALL BE 2 STOCK 7 PLY SHEETS (0°:45°:90°:45°:0°) OF G/PS PER 4, 1 FILLER PLY OF SGF/PS (181) CLOTH. SEE DETAIL

REV STATUS SH	REV STATUS LTR

REVISIONS ZONE	LTR	DESCRIPTION	DATE	APPROVED

PARTS LIST

QTY REQD -10	QTY REQD -19	QTY REQD -11	QTY REQD -4	QTY REQD -3	QTY REQD -2	QTY REQD -1	FIND NUMBER	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	MATERIAL AND SPECIFICATION	CODE IDENT NO	HT TR	FINISH	PT GR	REV LTR
	1							-21	DOUBLER	A4-4	5 13					
	1							-20	SKIN	A4-4	6 13					
1								-19	DOOR ASSY	A4-4	13					
				1				-18	SKIN - RH	B2-2	6 13					
					1			-17	SKIN - LH	B3-2	6 13					
						1		-16	OPP -15 RH	A7-3	MAKE FROM AND 10133-1404 L 7075-0 AL .188×1.5×1.5×8.1		T6	11		
						1		-15	ANGLE LH	A7-3	MAKE FROM AND 10133-1404 L 7075-0 AL .188×1.5×1.5×8.1		T6	11		
		2						-14	CAP-LOWER	B2-3	5 13					
		1						-13	CAP-UPPER	B2-3	6 13					
		1						-12	CORE	B2-3	3 13 15					
				1	1			-11	HC RING ASSY	B2-3	9 13					
—								-10	DOOR INSTALLATION	C4-4	2 16					
				1	1			-9	DOUBLER	A4-3	5 13					
				1				-8	DOUBLER	D2-2	5 13					
			1					-7	OPP -6	A6-3	7 13					
			1					-6	FLANGE	A6-3	7 13					
			2					-5	FLANGE STRIP	A6-3	6 13					
				1	1			-4	STRINGER ASSY	A6-3	3					
						1		-3	SKIN ASSY RH	B2-2	13 10 14					
						1		-2	SKIN ASSY LH	B3-2	13 10 14					
						—		-1	SKIN INSTALLATION		2					

DIMENSIONING & TOLERANCING PER ANSI Y14.5	
UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES	
TOLERANCES: ANGLES ± 1°	
DECIMALS .XX ± .03	
DECIMALS .XXX ± .015	
CONTRACT NUMBER	
DRN	K W OSBORNE 3-18-75
CHK	
ENGR	J.H LAAKSO 3-21-75

THE BOEING COMPANY
CORPORATE OFFICES SEATTLE, WASHINGTON

SKIN INSTALLATION-CENTERBODY SECTION

SIZE J | CODE IDENT NO | SK031275K0

SCALE NOTED | SH 1 OF 4

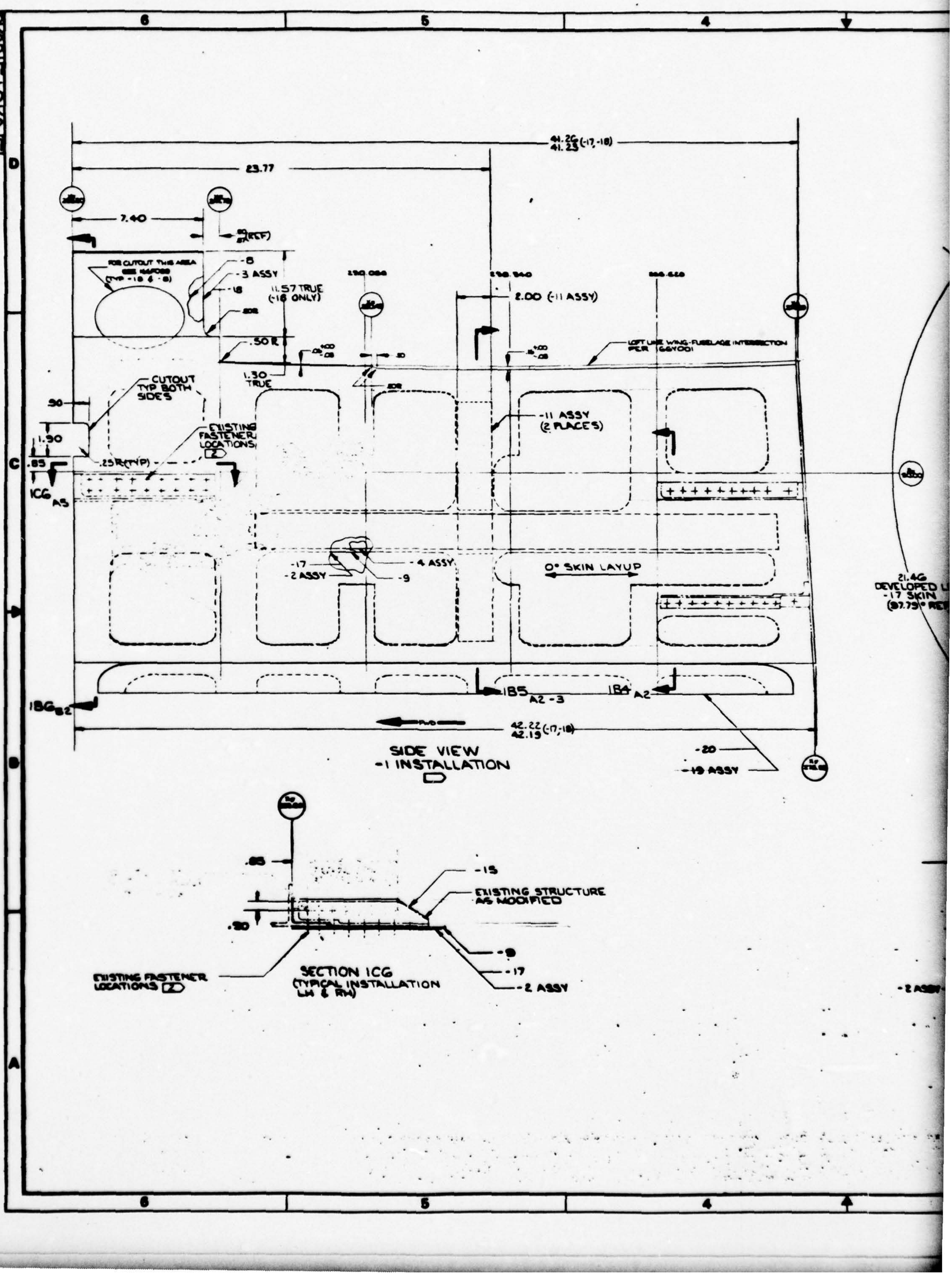
APPLICATION | FASTENER CODES

2-5560

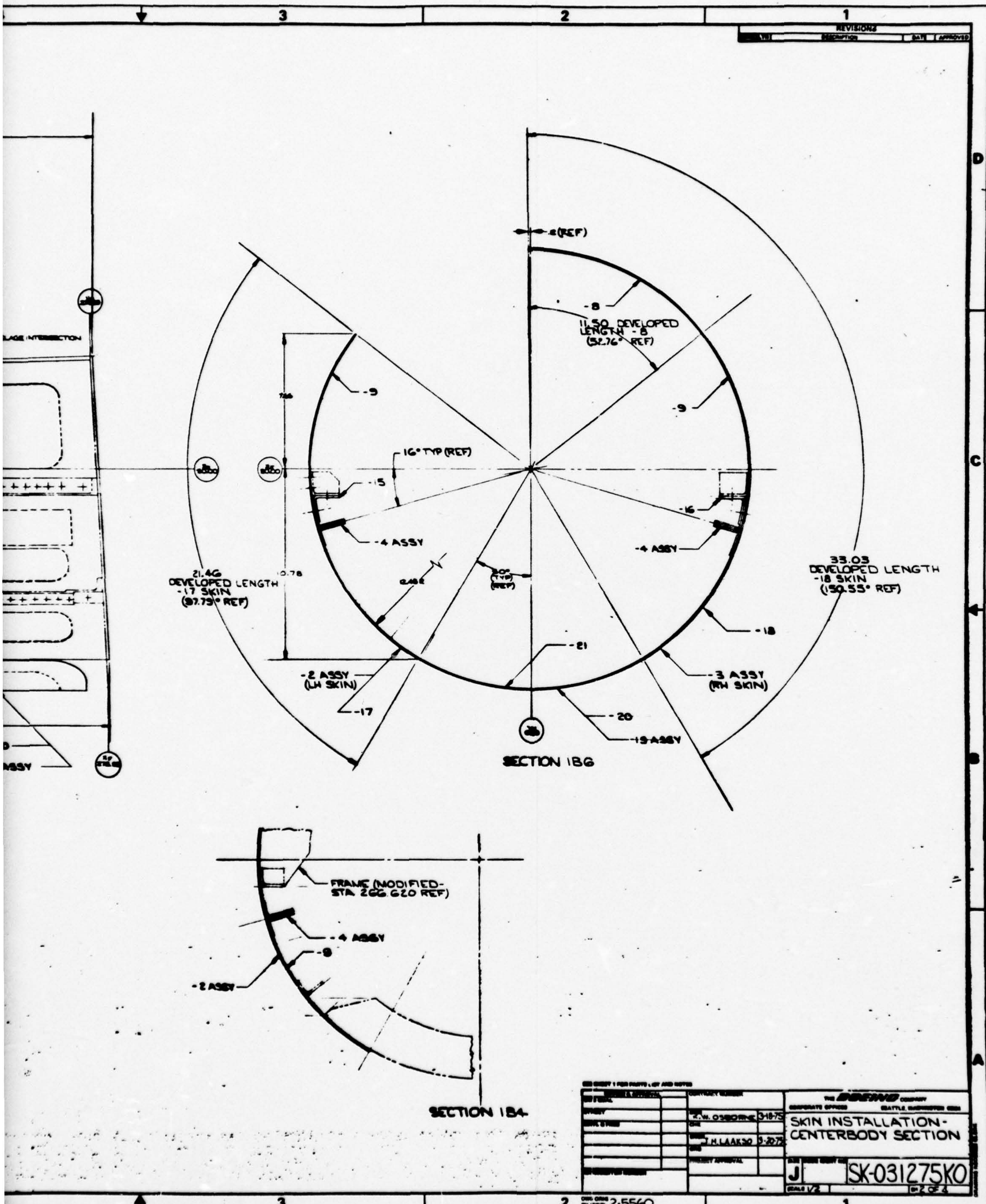
SK031275K0

A-2

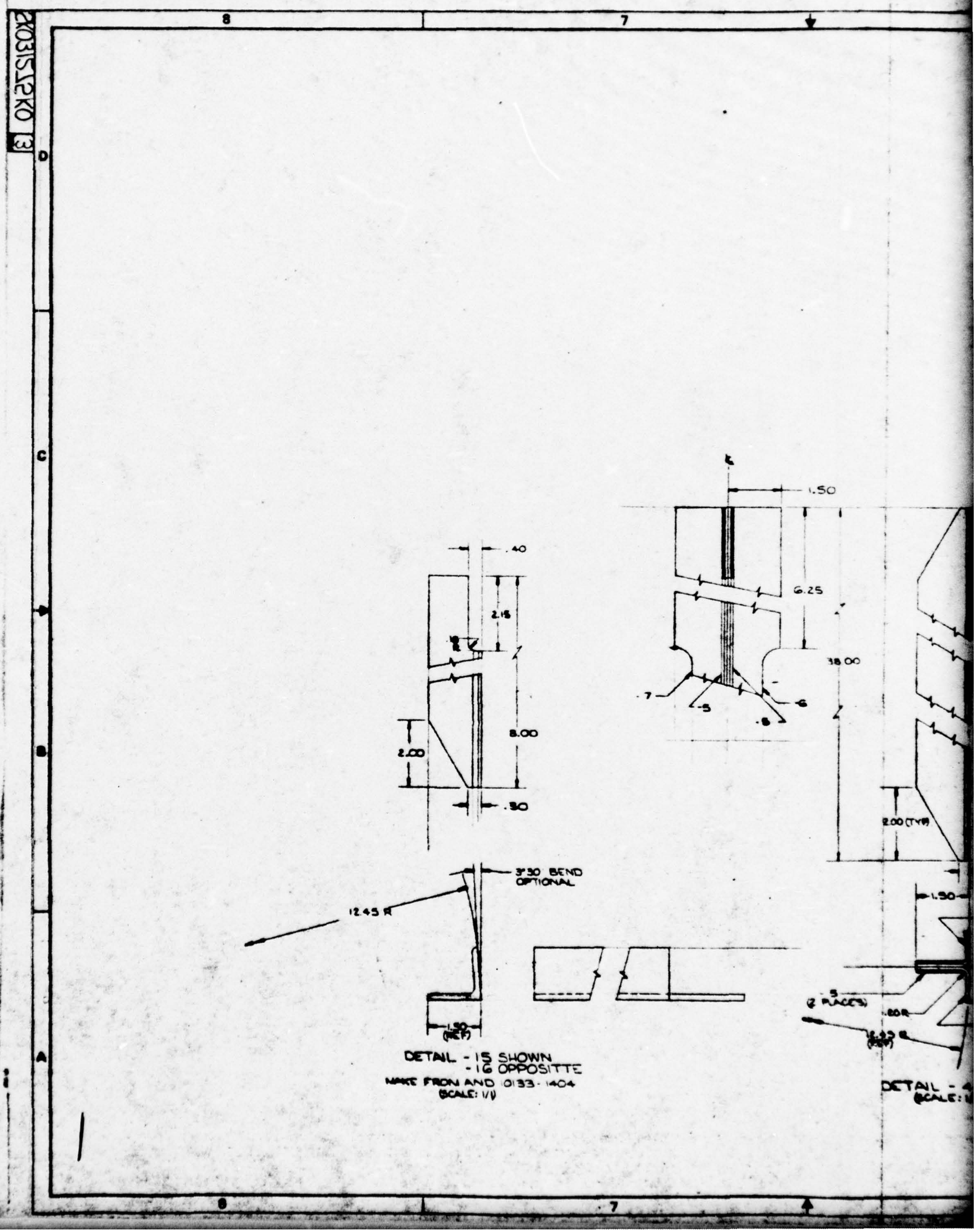
2



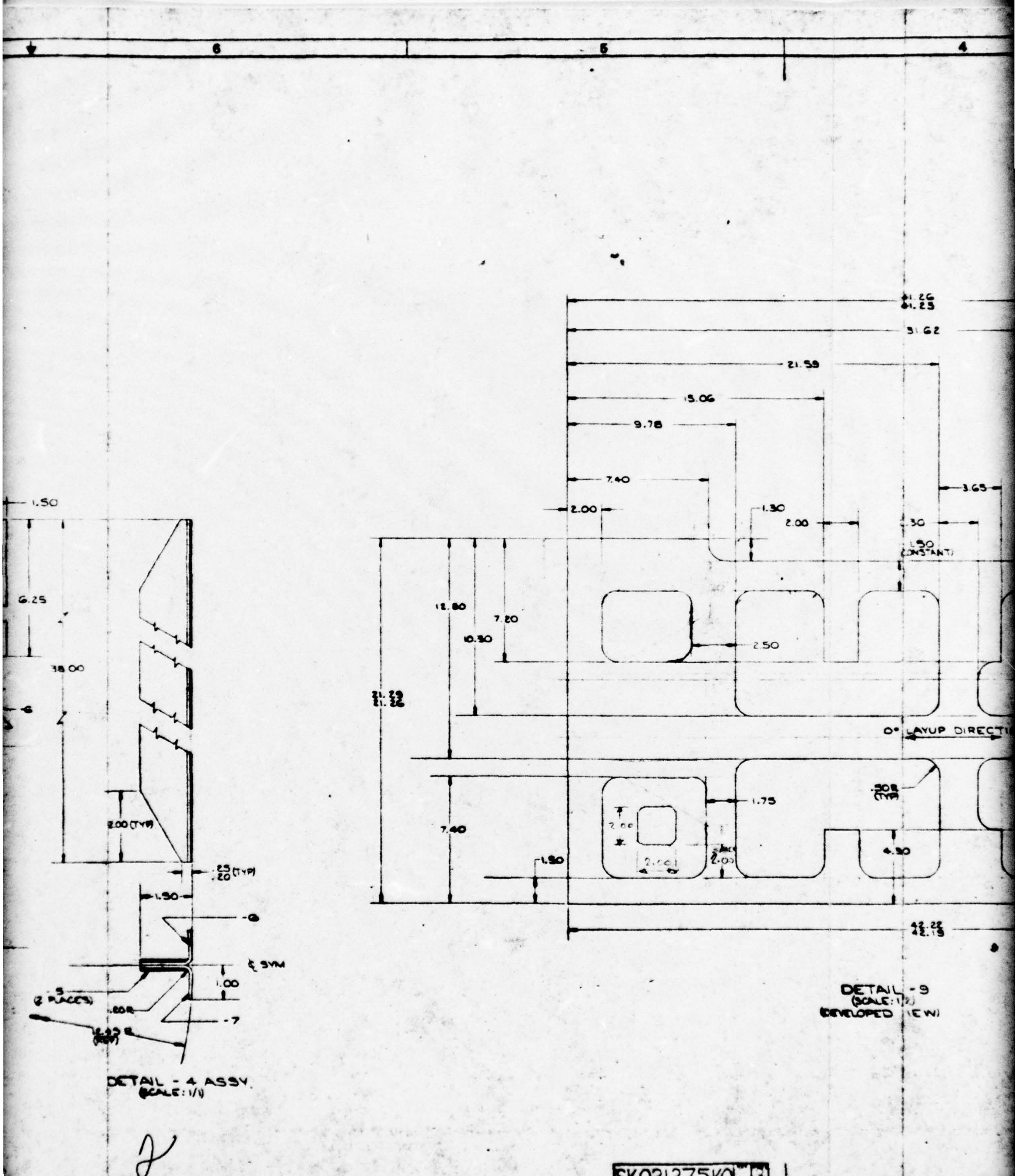
SIDE VIEW
-1 INSTALLATION
SECTION 1CG
(TYPICAL INSTALLATION LH & RH)
41.26 / 41.23 (-17,-18)
23.77
7.40
FOR CUTOUT THIS AREA
-8
-3 ASSY
-18
11.57 TRUE (-18 ONLY)
2.00 (-11 ASSY)
.50 R
1.30 TRUE
LOFT LINE WING-FUSELAGE INTERSECTION
CUTOUT TYP BOTH SIDES
EXISTING FASTENER LOCATIONS
.25 R (TYP)
-11 ASSY (2 PLACES)
1.90
.85
1CG
-17
-2 ASSY
-4 ASSY
-9
0° SKIN LAYUP
21.46
DEVELOPED L
-17 SKIN
1B6
1B5 A2 -3
1B4 A2
42.22 / 42.19 (-17,-18)
-20
-19 ASSY
.85
-15
EXISTING STRUCTURE AS MODIFIED
.30
-9
-17
-2 ASSY
EXISTING FASTENER LOCATIONS

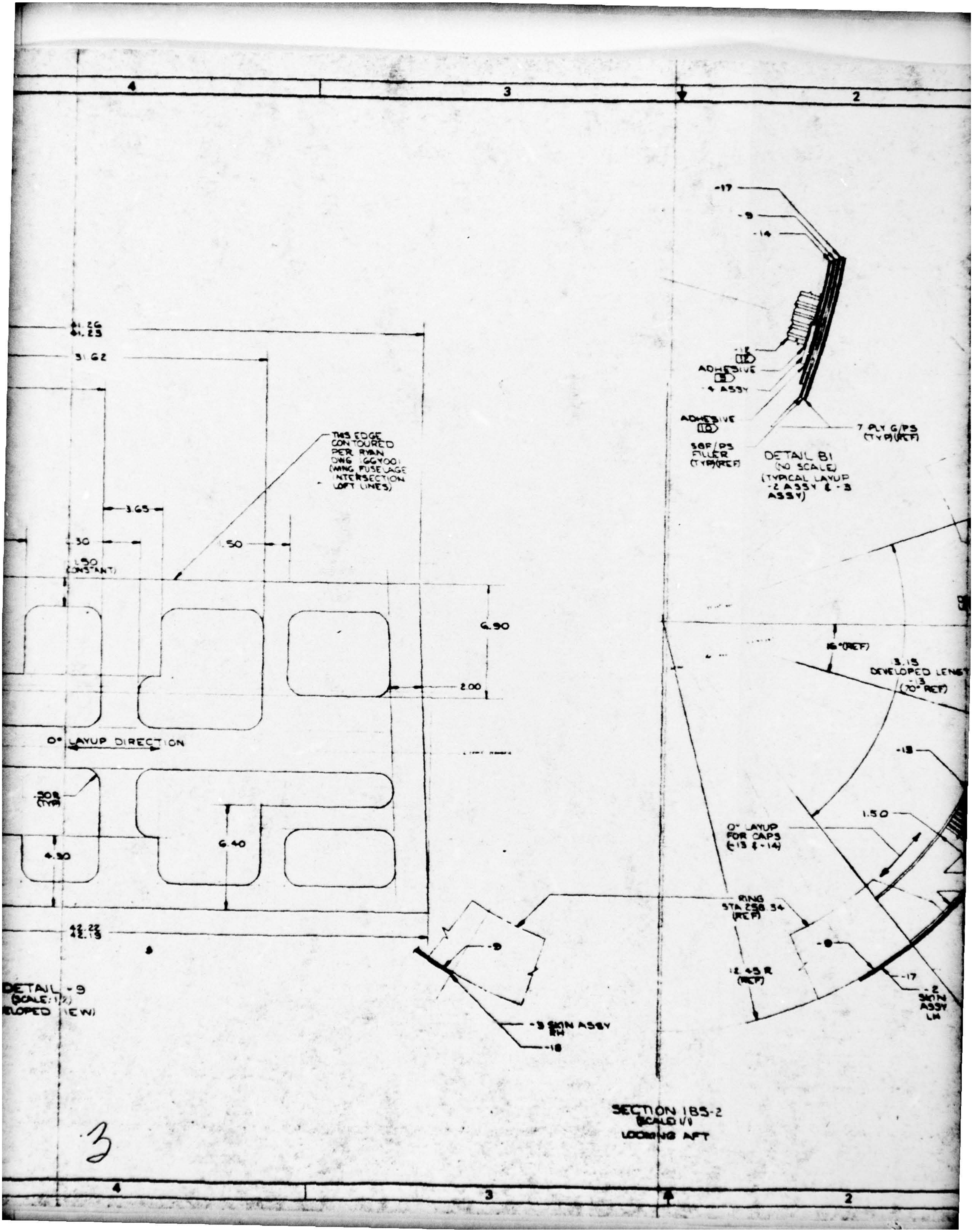


REVISIONS
.2 (REF)
-8
11.50 DEVELOPED LENGTH -8 (52.76° REF)
-9
-9
16° TYP (REF)
-15
-16
-4 ASSY
-4 ASSY
21.46 DEVELOPED LENGTH -17 SKIN (97.79° REF)
10.78
20° TYP (REF)
33.03 DEVELOPED LENGTH -18 SKIN (150.55° REF)
-18
-21
-2 ASSY (LH SKIN)
-3 ASSY (RH SKIN)
-17
-20
-19 ASSY
SECTION 1B6
FRAME (MODIFIED-STA 266.620 REF)
-4 ASSY
-9
-2 ASSY
SECTION 1B4
J.W. OSBORNE 3-18-75
J.M. LAAKSO 3-20-75
SKIN INSTALLATION-CENTERBODY SECTION
J
SK-031275KO
SCALE 1/2
SH 2 OF 4
SK-031275KO 12
2-5560



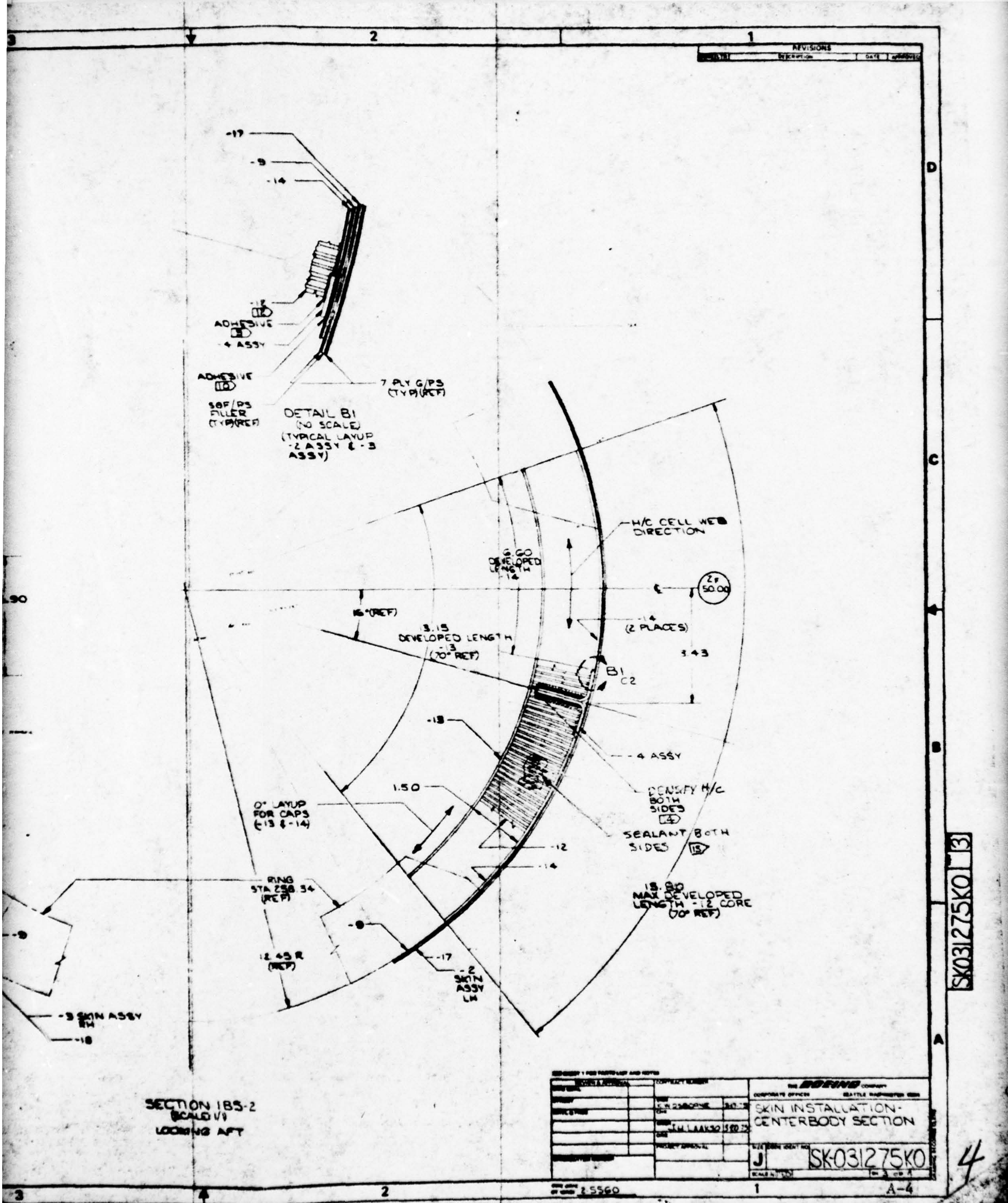
.40
2.15
8.00
2.00
.30
1.50
6.25
38.00
-7
-5
-6
-8
2.00 (TYP)
3°30 BEND
OPTIONAL
12.45 R
1.50
(REF)
DETAIL - 15 SHOWN
- 16 OPPOSITTE
MAKE FROM AND 10133-1404
(SCALE: 1/1)
-5
(2 PLACES)
DETAIL - 4


6
5
4
1.50
6.25
38.00
2.00 (TYP)
.25
.20 (TYP)
1.50
-8
℄ SYM
1.00
-5
(2 PLACES)
-7
DETAIL - 4 ASSY.
(SCALE: 1/1)
21.26
21.25
31.62
21.59
15.06
9.78
7.40
2.00
1.30
2.00
3.65
.30
.50
(CONSTANT)
12.80
7.20
10.30
2.50
21.29
21.26
0° LAYUP DIRECTI
.308
(TYP)
1.75
7.40
2.00
2.00
4.90
1.90
42.22
42.19
DETAIL - 9
(DEVELOPED VIEW)
SK031275K01
3



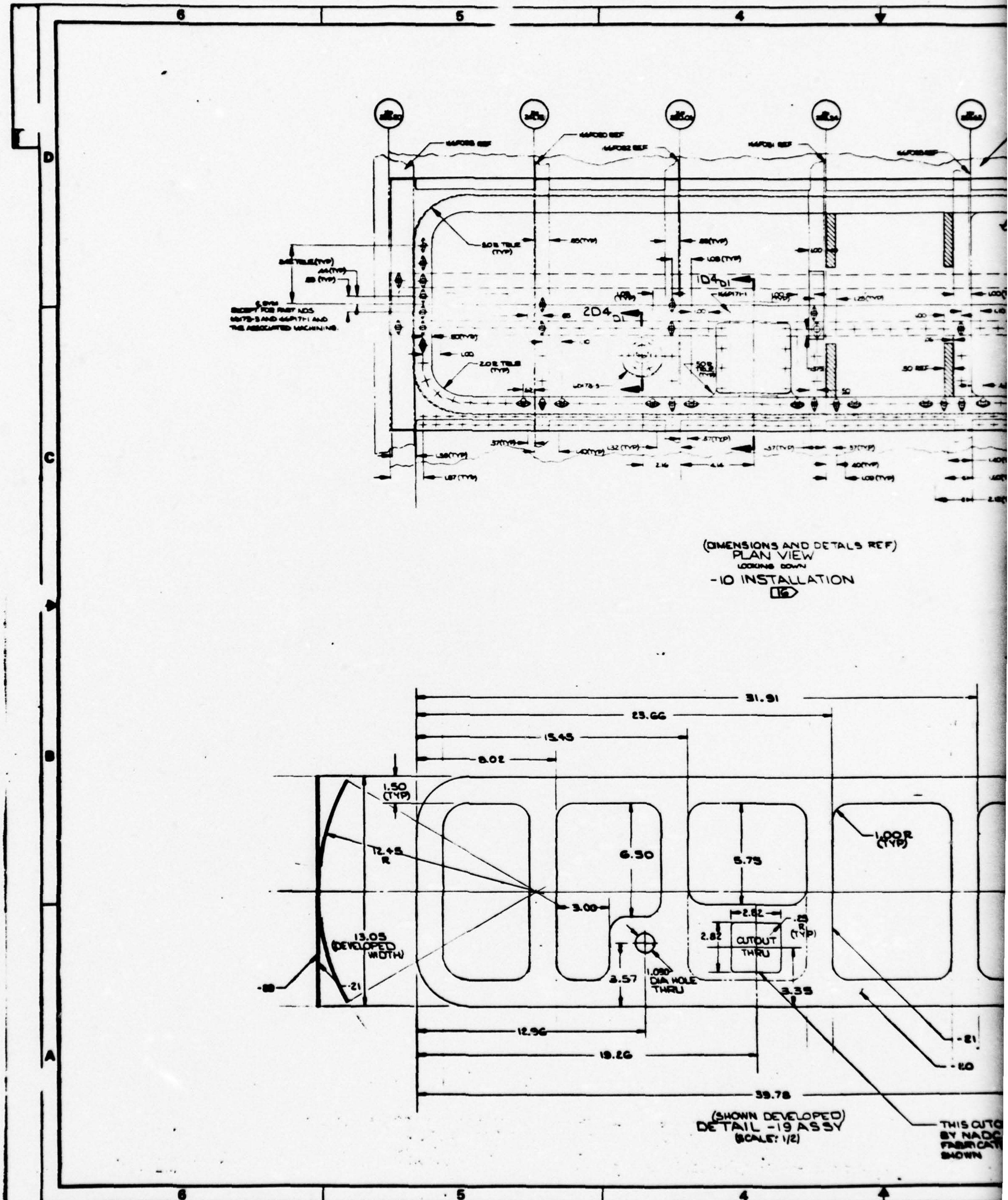
4
3
2
-17
-9
-14
-18
ADHESIVE
-4 ASSY
ADHESIVE
7 PLY G/PS (TYP)(REF)
SBF/PS FILLER (TYP)(REF)
DETAIL B1
(NO SCALE)
(TYPICAL LAYUP -2 ASSY & -3 ASSY)
41.26
41.23
31.62
THIS EDGE CONTOURED PER RYAN DWG 166Y001 (WING FUSELAGE INTERSECTION LOFT LINES)
3.65
.30
1.50
1.50 CONSTANT)
6.90
2.00
16° (REF)
13.15
DEVELOPED LENGTH
-3
(70° REF)
0° LAYUP DIRECTION
.508 (TYP)
6.40
4.90
-13
1.50
0° LAYUP FOR CAPS (-13 & -14)
RING STA 258.34 (REF)
42.22
42.19
-9
-9
12.45 R (REF)
-17
-2 SKIN ASSY LH
DETAIL -9
(SCALE: 1/2)
DEVELOPED VIEW
-3 SKIN ASSY RH
-18
SECTION 1BS-2
(SCALE 1/1)
LOOKING AFT
4
3
2

3



REVISIONS
-17
-9
-14
-12
ADHESIVE
-4 ASSY
ADHESIVE
SGF/PS FILLER (TYP)(REF)
7 PLY G/PS (TYP)(REF)
DETAIL B1 (NO SCALE) (TYPICAL LAYUP -2 ASSY & -3 ASSY)
H/C CELL WEB DIRECTION
6.60 DEVELOPED LENGTH -14
50.00
16° (REF)
13.15 DEVELOPED LENGTH -13 (70° REF)
-14 (2 PLACES)
3.43
B1 C2
-13
-4 ASSY
DENSIFY H/C BOTH SIDES
1.50
0° LAYUP FOR CAPS (-13 & -14)
SEALANT BOTH SIDES
-12
-14
15.80 MAX DEVELOPED LENGTH -12 CORE (70° REF)
RING STA 258.34 (REF)
-9
12.45 R (REF)
-17
-2 SKIN ASSY LH
-3 SKIN ASSY RH
-18
SECTION 185-2 (SCALE 1/1) LOOKING AFT
SKIN INSTALLATION-CENTERBODY SECTION
J
SK-031275K0
SK031275K0 13
2-5560
A-4
4

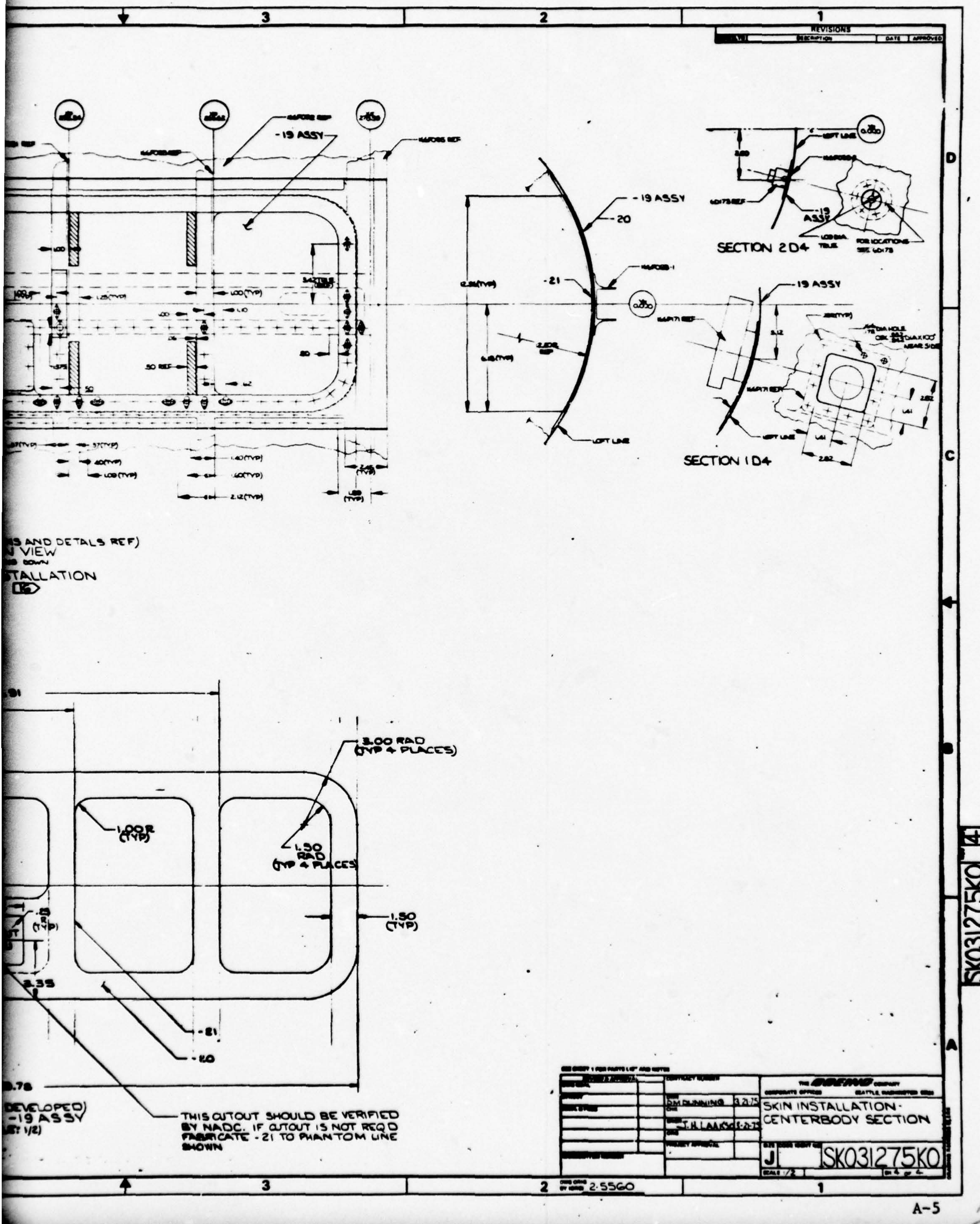
6 | 5 | 4



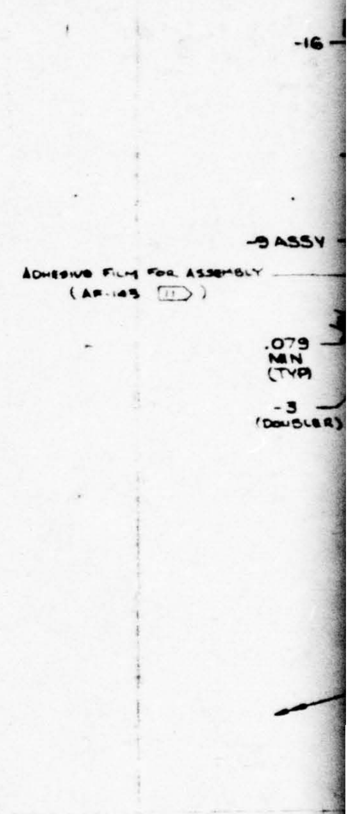
(DIMENSIONS AND DETAILS REF)
PLAN VIEW
LOOKING DOWN
-10 INSTALLATION

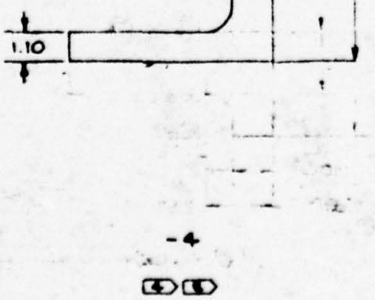
(SHOWN DEVELOPED)
DETAIL -19 ASSY
(SCALE: 1/2)

6 | 5 | 4



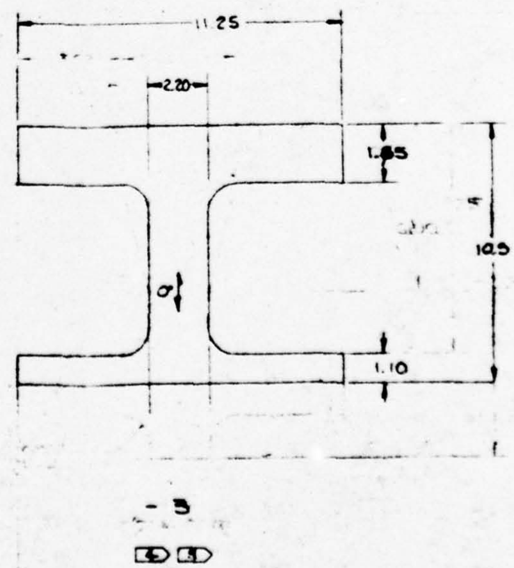
REVISIONS
-19 ASSY
-20
-21
LOFT LINE
SECTION 2 D4
SECTION 1 D4
FOR LOCATIONS SEE 60173
(S AND DETALS REF)
N VIEW
STALLATION
3.00 RAD (TYP 4 PLACES)
1.00 R (TYP)
1.50 RAD (TYP 4 PLACES)
1.50 (TYP)
.25 R (TYP)
2.35
-21
-20
(DEVELOPED)
-19 ASSY
THIS CUTOUT SHOULD BE VERIFIED BY NADC. IF CUTOUT IS NOT REQ'D FABRICATE -21 TO PHANTOM LINE SHOWN
SKIN INSTALLATION - CENTERBODY SECTION
SEATTLE, WASHINGTON
D M DUNNING 3-21-75
T.H. LAAKSO 3-21-75
J
SK031275K0
2-5560
SK031275K0 14





-4

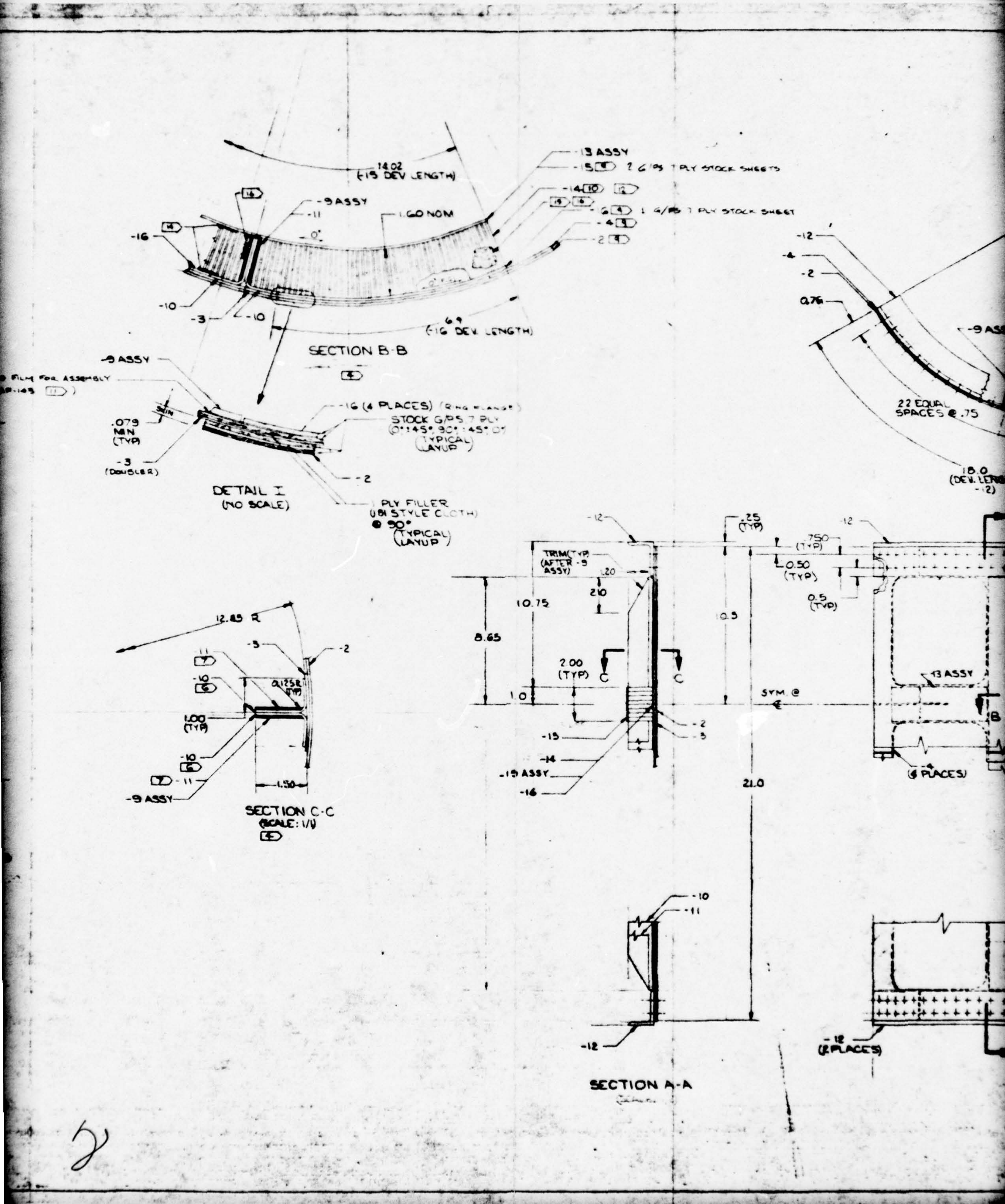
4 5



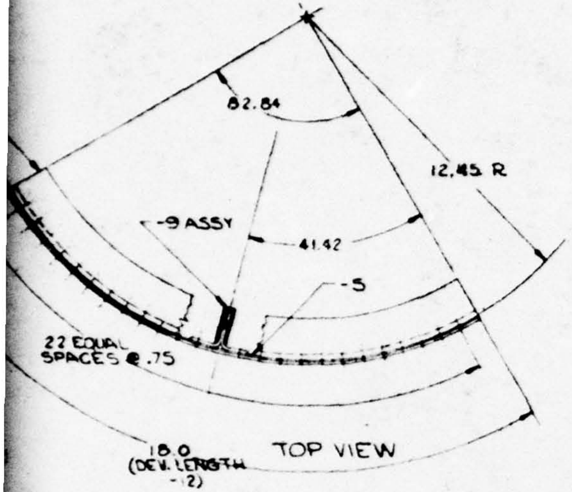
-3

4 5

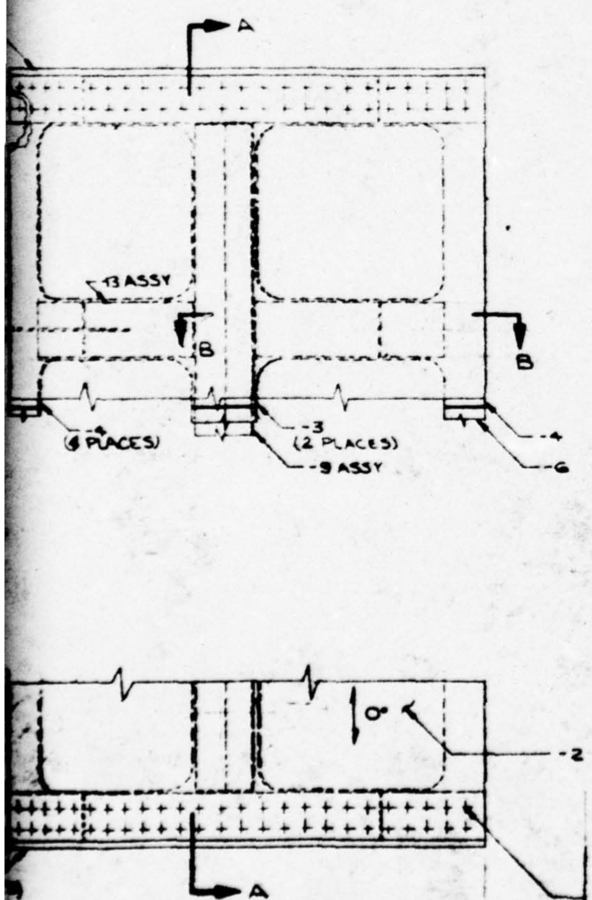
NOTE: DOUBLER DETAILS -3 & -4 ARE OPTIONAL. DOUBLER MAY BE FABRICATED IN ONE PIECE



14.02
(-15 DEV LENGTH)
-13 ASSY
-15 2 G/PS 7 PLY STOCK SHEETS
-14
-16 1 G/PS 7 PLY STOCK SHEET
-4
-2
-9 ASSY
-11
1.60 NOM
-16
-10
-3
6.9
(-16 DEV. LENGTH)
SECTION B-B
-9 ASSY
.079
MIN
(TYP)
-3
(DOUBLER)
-16 (4 PLACES) (RING FLANGE)
STOCK G/PS 7 PLY
(TYPICAL LAYUP)
-2
1 PLY FILLER
181 STYLE CLOTH
@ 90°
(TYPICAL LAYUP)
DETAIL I
(NO SCALE)
-12
-4
-2
0.76
-9 ASSY
22 EQUAL SPACES @ .75
18.0
(DEV. LENGTH -12)
12.85 R
-3
-2
-11
-10
0.125R (TYP)
1.00
(TYP)
-10
-11
-9 ASSY
1.50
SECTION C-C
(SCALE: 1/1)
.25
(TYP)
TRIM (TYP)
(AFTER -9 ASSY)
1.20
2.10
10.75
8.65
2.00
(TYP)
C
C
1.0
10.5
.750
(TYP)
0.50
(TYP)
0.5
(TYP)
SYM
-13 ASSY
B
-15
-14
-15 ASSY
-16
-2
-3
21.0
-4
(6 PLACES)
-10
-11
-12
-12
(2 PLACES)
SECTION A-A



TOP VIEW



.203 / .199 DIA HOLE
CSK 100° × .400 / .390 DIA
NEAR SIDE
3/16 DIA BLIND FASTENER (100° HD)
CLASS 1 STEEL
(BACB30LB -XLR RECOMMENDED OR BACB30AY -DK)
(12 PLACES)

-1 ASSEMBLY
FRONT VIEW

16 APPLY BMS 5-26 POLYSULFIDE SEALANT TO SEAL ALL GAPS, .025 ... AND H/C SIDES AND ENDS

15 APPLY RIGID FOAM IN CELLS (BMS 5-90) TO REINFORCE H/C SIDES & ... 2½ CELLS DEEP FOR SIDES, 10 CELLS DEEP FOR ENDS

14 AF-143 EPOXY FILM ADHESIVE TYPE .01 PSF PER BMS 5-104.

13 NOMEX BMS 8-124, CLASS IV, TYPE I, HEAT FORMED PER BAC 52...

12 MILL H/C CORE TO FIT STOP AT STRINGER ANGLE LEG AND ...

11 AF-143 EPOXY FILM ADHESIVE, 0.05 PSF; SCOTCHBRITE / ALCOHOL C... CURE 1 HR @ 325-335°F, 4-25 PSIG.

10 HRP-10 3/16 CELL HEXCELL HONEYCOMB

9 G/PS TYPE II - PRELIMINARY MATL SPEC. UNIDIRECTI... GRAPHITE PREIMPREGNATED TAPE THERMOPLASTIC ... JUNE 1974. PS MATL IS UNION CARBIDE P-1700.

7 FLANGE STRIPS (-11) SHALL BE 15 PLIES UNIDIRECTIONA...

6 STRINGERS (-10) SHALL BE 9 PLIES SGF/PS (181) CLOTH ... 45° LAYUP ORIENTATION.

5 SKIN & DOUBLER LAMINATES SHALL BE 2 STOCK 7... SHEETS (0° ±45° 90° ±45° 0°) OF G/PS TYPE II. 1 FILL... SGF/PS (181) CLOTH, SEE DET. I

4 FABRICATE LAMINATE SUB ASSY PER FAB PLAN CS ...

3 PAINT WITH ZINC CHROMATE PRIMER

2 HT-TR TO T6 COND AFTER FORMING

1 MAKE FROM AND 10134 - 2008 EXT L, 7075 AL...

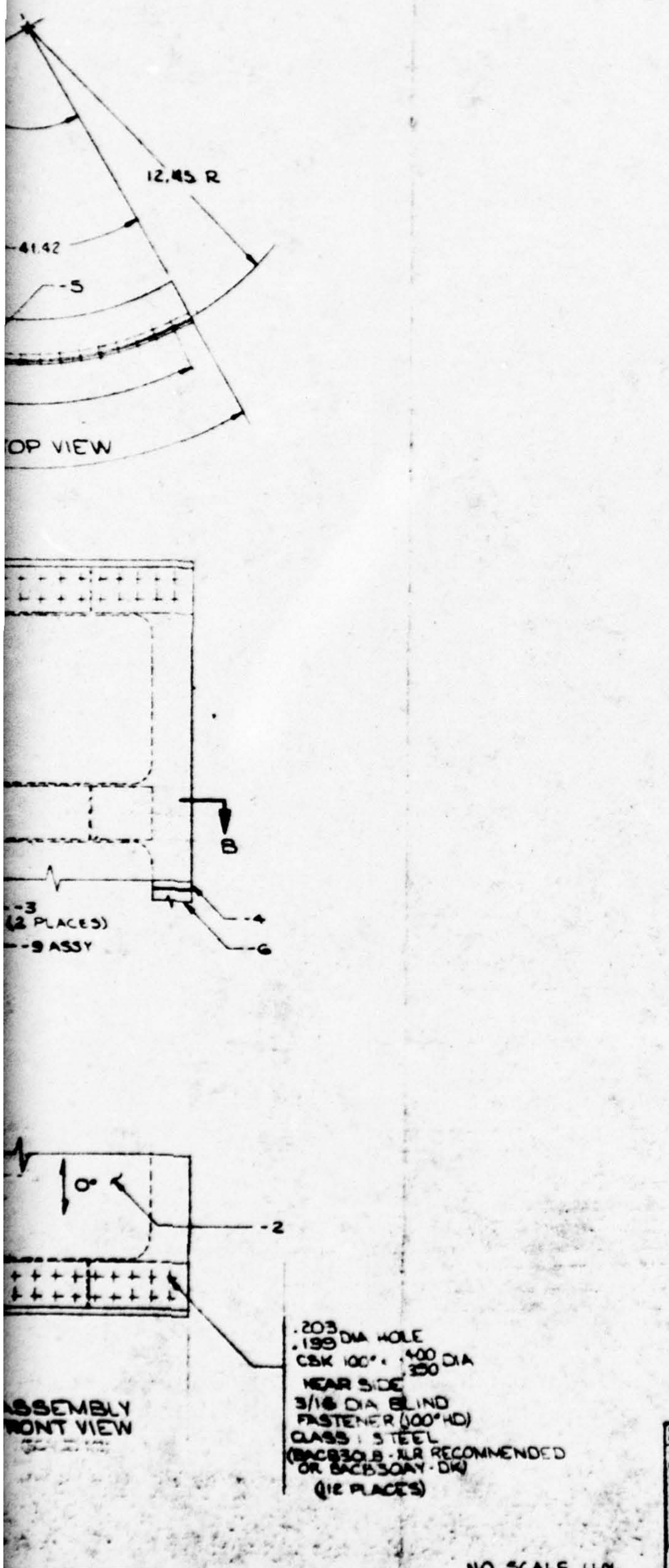
SEE SHEET 1 FOR SEPARATE ...

NO SCALE U.N.

SUBCOMPONE...

SK22...

3



16 APPLY BMS 5-26 POLYSULFIDE SEALANT TO SEAL ALL GAPS, .025 IN MIN FILLET AND H/C SIDES AND ENDS

15 APPLY RIGID FOAM IN CELLS (BMS 5-90) TO REINFORCE H/C SIDES & ENDS 2½ CELLS DEEP FOR SIDES, 10 CELLS DEEP FOR ENDS

14 AF-143 EPOXY FILM ADHESIVE TYPE CI PSF PER BMS 5-104

13 NOMEX BMS 8-124, CLASS IV, TYPE I, HEAT FORMED PER BAC 5201

12 MILL H/C CORE TO FIT STOP AT STRINGER ANGLE LEG AND G/PS SEC

11 AF-143 EPOXY FILM ADHESIVE, 0.05 PSF; SCOTCHBRITE / ALCOHOL CLEAN, CURE 1 HR @ 325-335°F, 4-25 PSIG.

10 HRP-10 3/16 CELL HEXCELL HONEYCOMB

9 G/PS TYPE II - PRELIMINARY MATL SPEC. UNIDIRECTIONAL GRAPHITE PREIMPREGNATED TAPE THERMOPLASTIC MATRIX JUNE 1974. PS MATL IS UNION CARBIDE P-1700.

7 FLANGE STRIPS (-11) SHALL BE 15 PLIES UNIDIRECTIONAL G/PS, TYPE II

6 STRINGERS (-10) SHALL BE 9 PLIES SGF/PS (181) CLOTH WITH 45° LAYUP ORIENTATION.

5 SKIN & DOUBLER LAMINATES SHALL BE 2 STOCK 7 PLY SHEETS (0°, ±45°, 90°, ±45°, 0°) OF G/PS TYPE II, 1 FILLER PLY SGF/PS (181) CLOTH, SEE DET. I

4 FABRICATE LAMINATE SUB ASSY PER FAB PLAN CS 25540 FP-1

3 PAINT WITH ZINC CHROMATE PRIMER

2 HT-TR TO T6 COND AFTER FORMING

1 MAKE FROM AND10134-2008 EXT L, 7075 ALUM

REV 4-29-75

SEE SHEET 1 FOR SEPARATE PARTS LIST

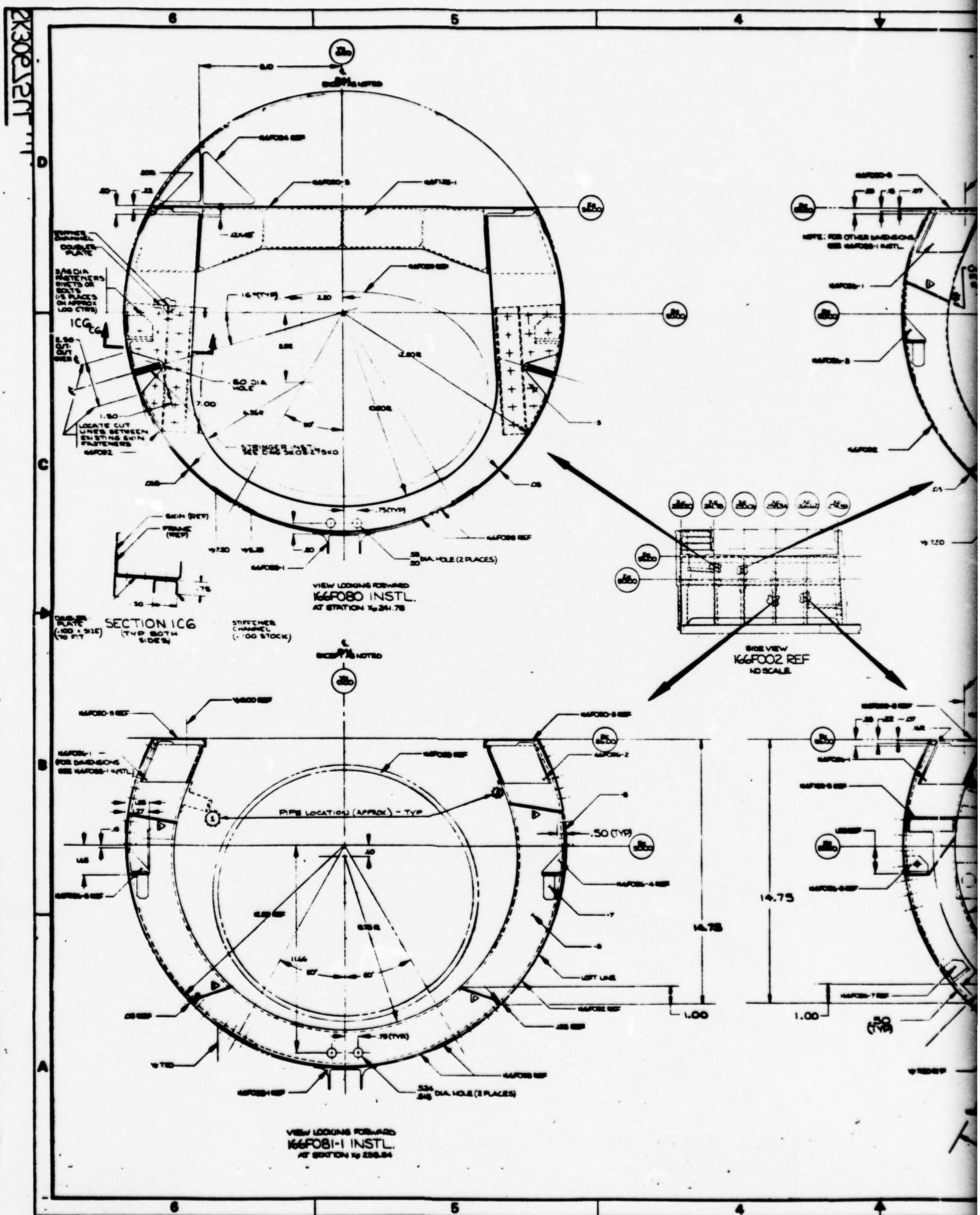
THE BOEING COMPANY

E. W. OSBORNE 2-12-75

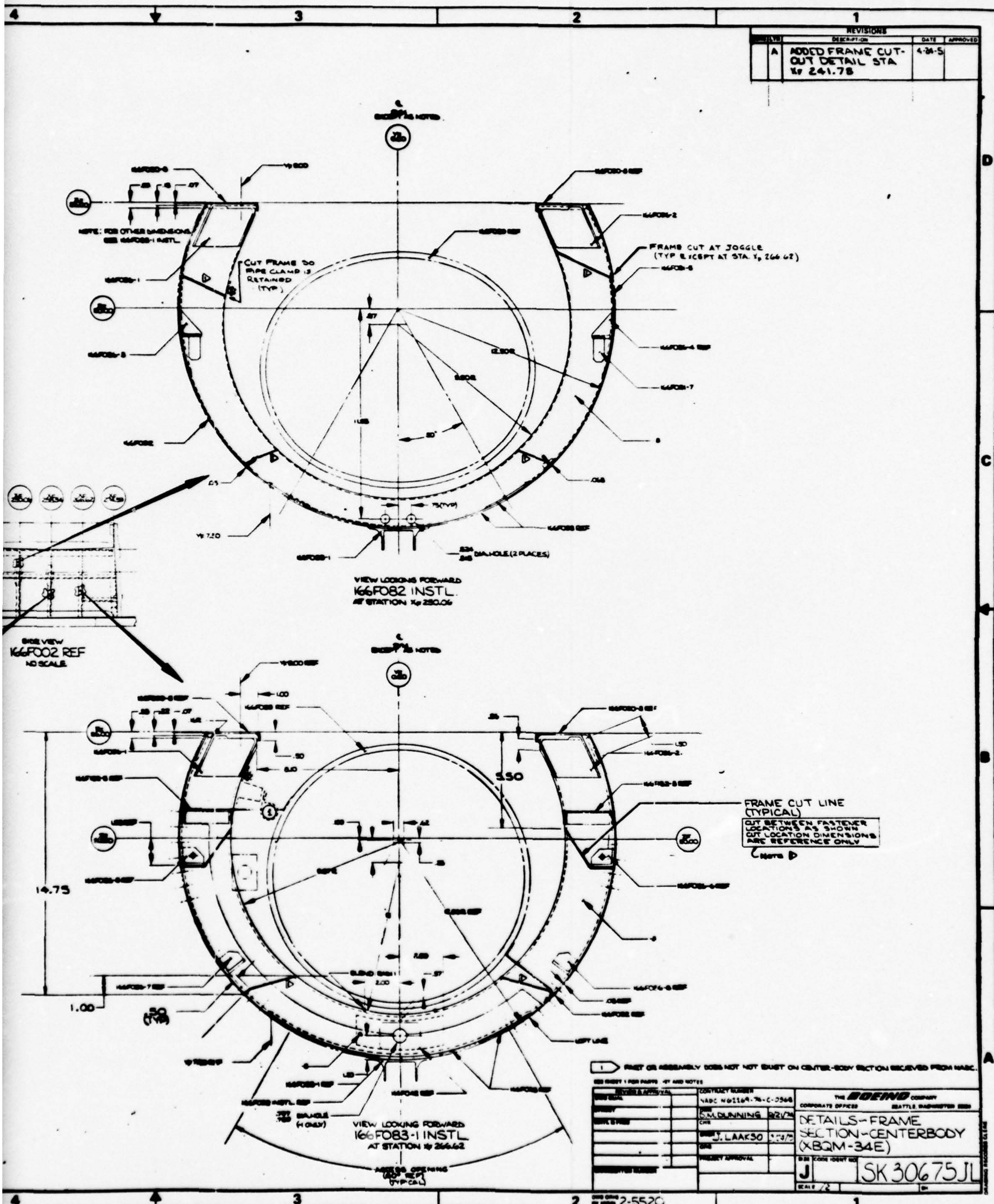
J. H. LAAKSO 2-28-75

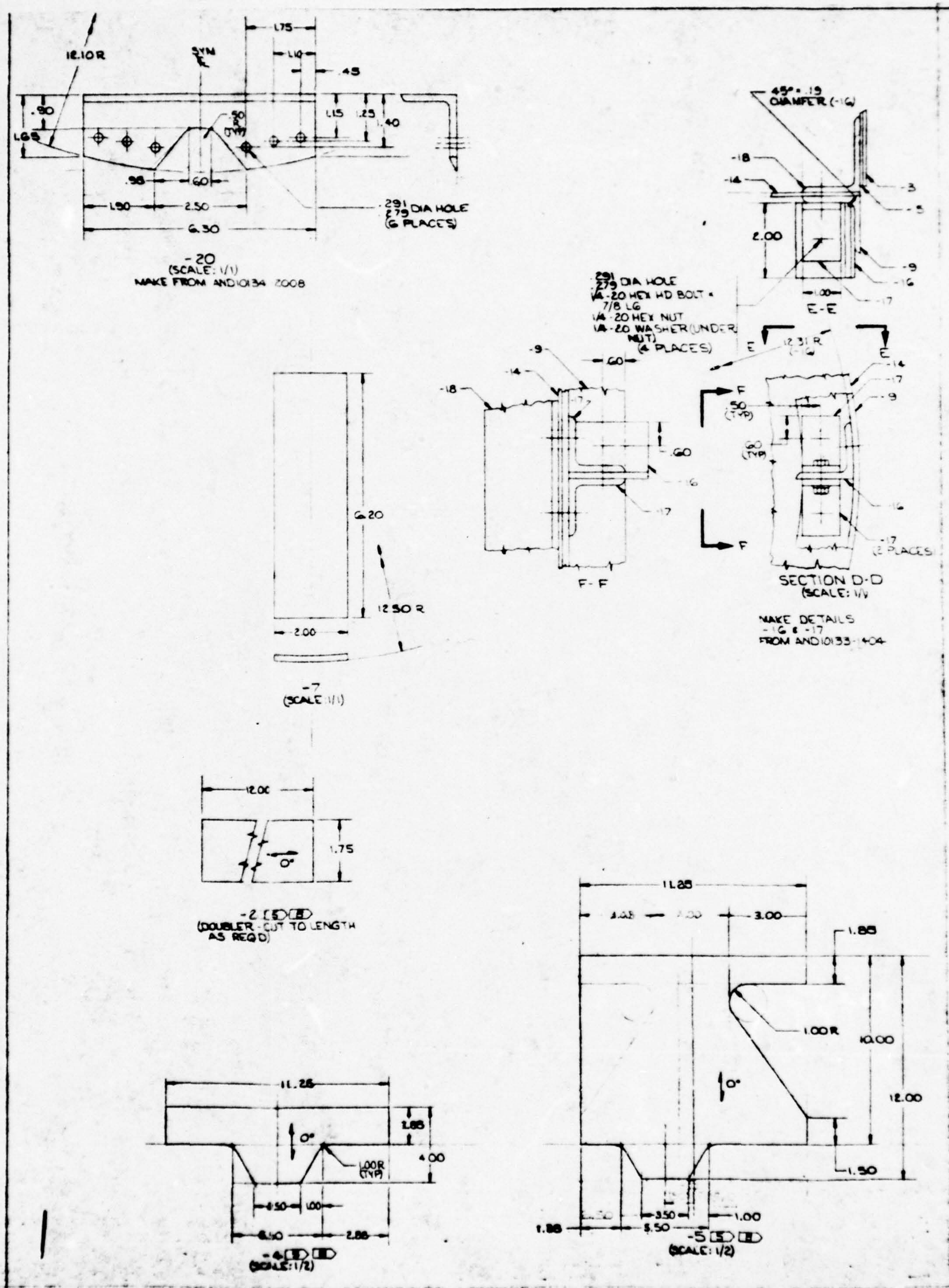
SUBCOMPONENT No 2

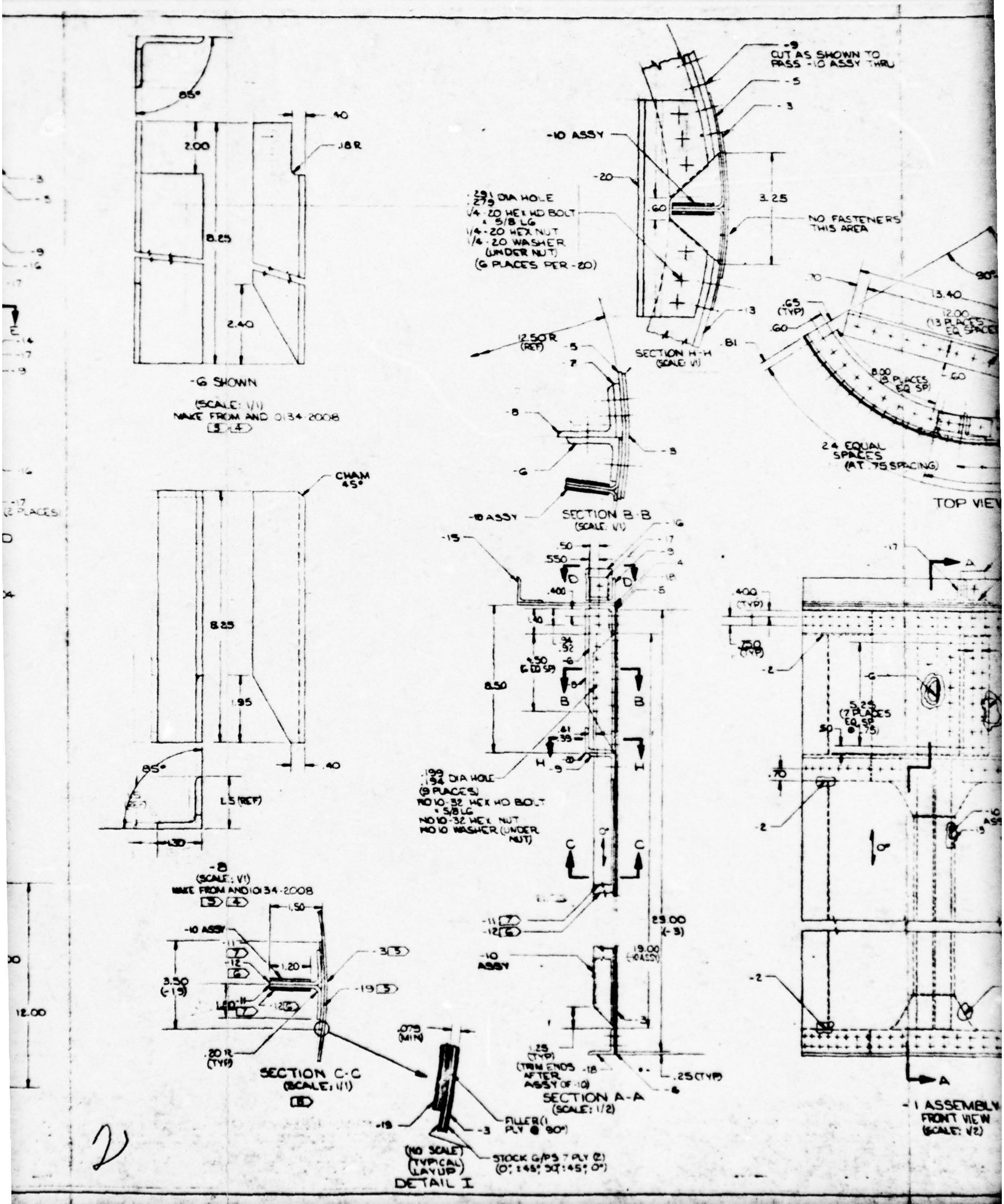
SK22175K0

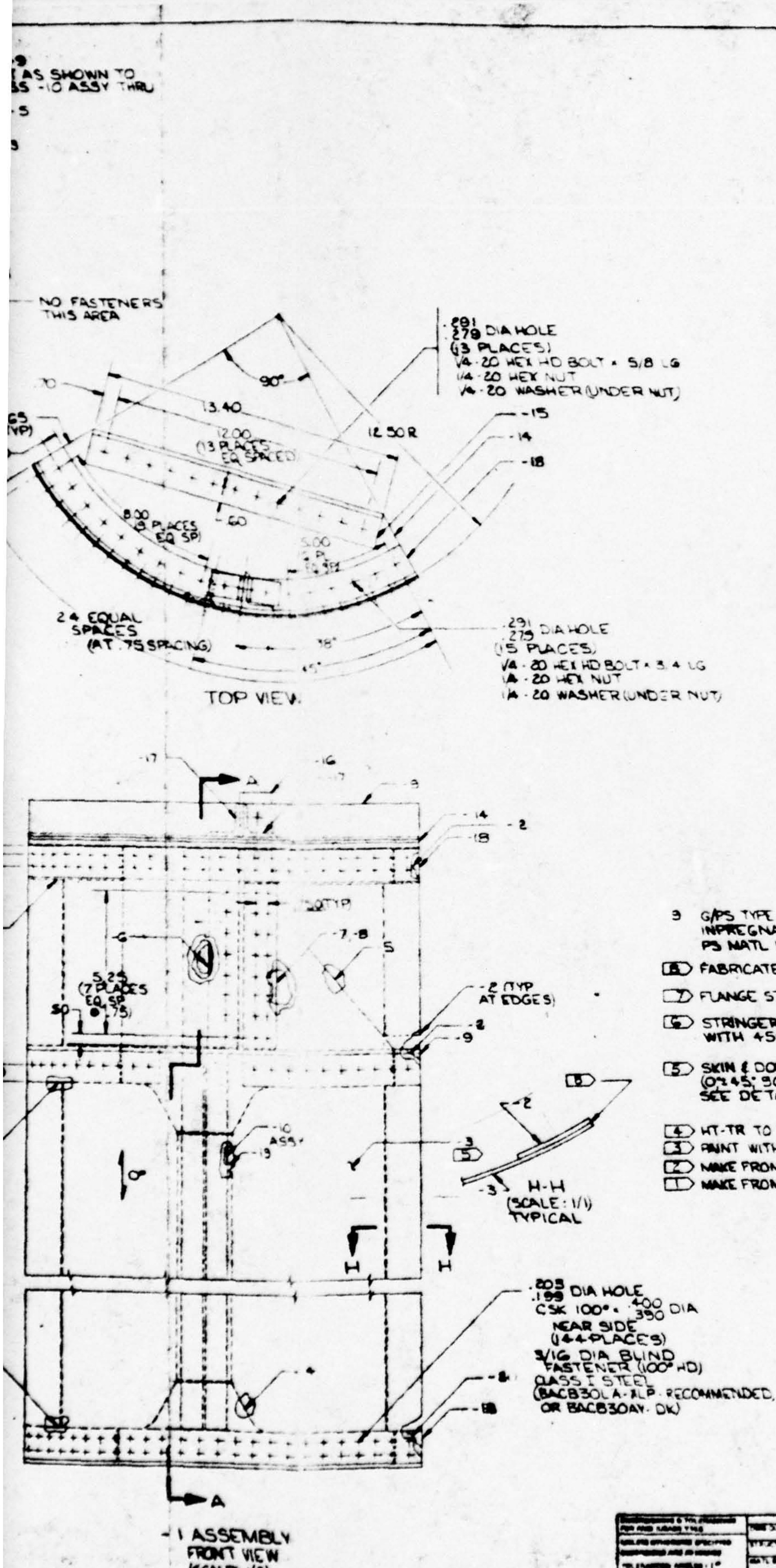


6
5
4
D
C
B
A
LOCATE CUT LINES BETWEEN EXISTING SKIN FASTENERS
7.00
1.50
SKIN (REF)
FRAME (REF)
SECTION 1C6
(TYP BOTH SIDES)
VIEW LOOKING FORWARD
166F080 INSTL.
SIDE VIEW
166F002 REF
NO SCALE
PIPE LOCATION (APPROX) - TYP
.50 (TYP)
14.75
1.00
LOFT LINE
DIA. HOLE (2 PLACES)
VIEW LOOKING FORWARD
166F081-1 INSTL.


REVISIONS
DESCRIPTION
DATE
APPROVED
A
ADDED FRAME CUT-OUT DETAIL STA Yf 241.78
4-24-5
FRAME CUT AT JOGGLE (TYP EXCEPT AT STA. Yf 266.62)
CUT FRAME SO PIPE CLAMP IS RETAINED (TYP)
VIEW LOOKING FORWARD
166F082 INSTL.
AT STATION Yf 250.06
SIDE VIEW
166F002 REF
NO SCALE
FRAME CUT LINE (TYPICAL)
CUT BETWEEN FASTENER LOCATIONS AS SHOWN
CUT LOCATION DIMENSIONS ARE REFERENCE ONLY
5.50
14.75
1.00
VIEW LOOKING FORWARD
166F083-1 INSTL.
AT STATION Yf 266.62
ACCESS OPENING (TYPICAL)
CONTRACT NUMBER
NADC N62269-74-C-0368
S.M. DUNNING 9/21/74
J. LAAKSO
THE BOEING COMPANY
DETAILS - FRAME
SECTION - CENTERBODY
(XBQM-34E)
J
SK 30675JL
2-5520
SK30675JL(A)


12.10 R
SYM
℄
1.75
1.10
.45
.90
1.65
.20 (TYP)
1.15
1.25
1.40
.98
.60
1.90
2.50
6.30
.291
.279 DIA HOLE
(6 PLACES)
-20
(SCALE: 1/1)
MAKE FROM AND10134-2008
45° x .19
CHAMFER (-16)
-18
-14
-3
-5
2.00
1.00
-9
-16
-17
E-E
.291
.279 DIA HOLE
1/4-20 HEX HD BOLT x 7/8 LG
1/4-20 HEX NUT
1/4-20 WASHER (UNDER NUT)
(4 PLACES)
12.31 R (-16)
E
E
-9
-14
.60
-18
-17
.60
-16
-17
F-F
F
.50 (TYP)
.60 (TYP)
-14
-17
-9
-16
-17 (2 PLACES)
F
SECTION D-D
(SCALE: 1/1)
MAKE DETAILS -16 & -17 FROM AND10133-1404
6.20
12.50 R
2.00
-7
(SCALE: 1/1)
12.00
1.75
0°
-2
(DOUBLER - CUT TO LENGTH AS REQ'D)
11.28
3.00
1.88
1.00 R
10.00
12.00
0°
1.50
3.50
1.00
1.88
3.50
-5
(SCALE: 1/2)
11.28
2.88
0°
4.00
1.00 R (TYP)
3.50
1.00
6.40
2.88
-4
(SCALE: 1/2)


-6 SHOWN
(SCALE: 1/1)
MAKE FROM AND 0134-2008
SECTION H-H
(SCALE: 1/1)
CUT AS SHOWN TO PASS -10 ASSY THRU
NO FASTENERS THIS AREA
-10 ASSY
.291/.279 DIA HOLE
1/4-20 HEX HD BOLT × 5/8 LG
1/4-20 HEX NUT
1/4-20 WASHER (UNDER NUT)
(6 PLACES PER -20)
SECTION B-B
(SCALE: 1/1)
24 EQUAL SPACES (AT .75 SPACING)
TOP VIEW
CHAM 45°
-8
(SCALE: 1/1)
MAKE FROM AND 10134-2008
.199/.194 DIA HOLE
(9 PLACES)
NO 10-32 HEX HD BOLT × 5/8 LG
NO 10-32 HEX NUT
NO 10 WASHER (UNDER NUT)
SECTION C-C
(SCALE: 1/1)
FILLER (1 PLY @ 90°)
STOCK G/PS 7 PLY (2) (0°, ±45°, 90°, ±45°, 0°)
(NO SCALE)
(TYPICAL LAYUP)
DETAIL I
(TRIM ENDS AFTER ASSY OF -10)
SECTION A-A
(SCALE: 1/2)
-1 ASSEMBLY
FRONT VIEW
(SCALE: 1/2)



9 G/PS TYPE I. PRELIM. MATL SPEC. UNIDIRECTIONAL GRAPHITE PRE-IMPREGNATED TAPE THERMOPLASTIC MATRIX JUNE 1974
PS MATL IS UNION CARBIDE P-1700

[8] FABRICATE LAMINATE SUB ASSY PER FAB PLAN CS 25540

[7] FLANGE STRIPS -11 SHALL BE 15 PLIES UNIDIRECTIONAL G/PS TYPE II

[6] STRINGERS (-12) SHALL BE 9 PLIES SGF/PS (181) CLOTH WITH 45° LAY UP ORIENTATION.

[5] SKIN & DOUBLER LAMINATES SHALL BE 2 STOCK 7 PLY SHEETS (0°, ±45°, 90°, ±45°, 0°) OF G/PS TYPE I, 1 FILLER PLY SGF/PS (181) CLOTH. SEE DETAIL I.

[4] HT-TR TO T6 COND AFTER MACH OR FORMING
[3] PAINT WITH ZINC CHROMATE PRIMER
[2] MAKE FROM AND10134-2008 EXT. L, 7075 ALUM
[1] MAKE FROM AND10133-1404 EXT. L, 7075 ALUM

UPDATED - 2-27-75 KWO

SEE SHEET 1 FOR SEPARATE PARTS LIST

THE BOEING COMPANY

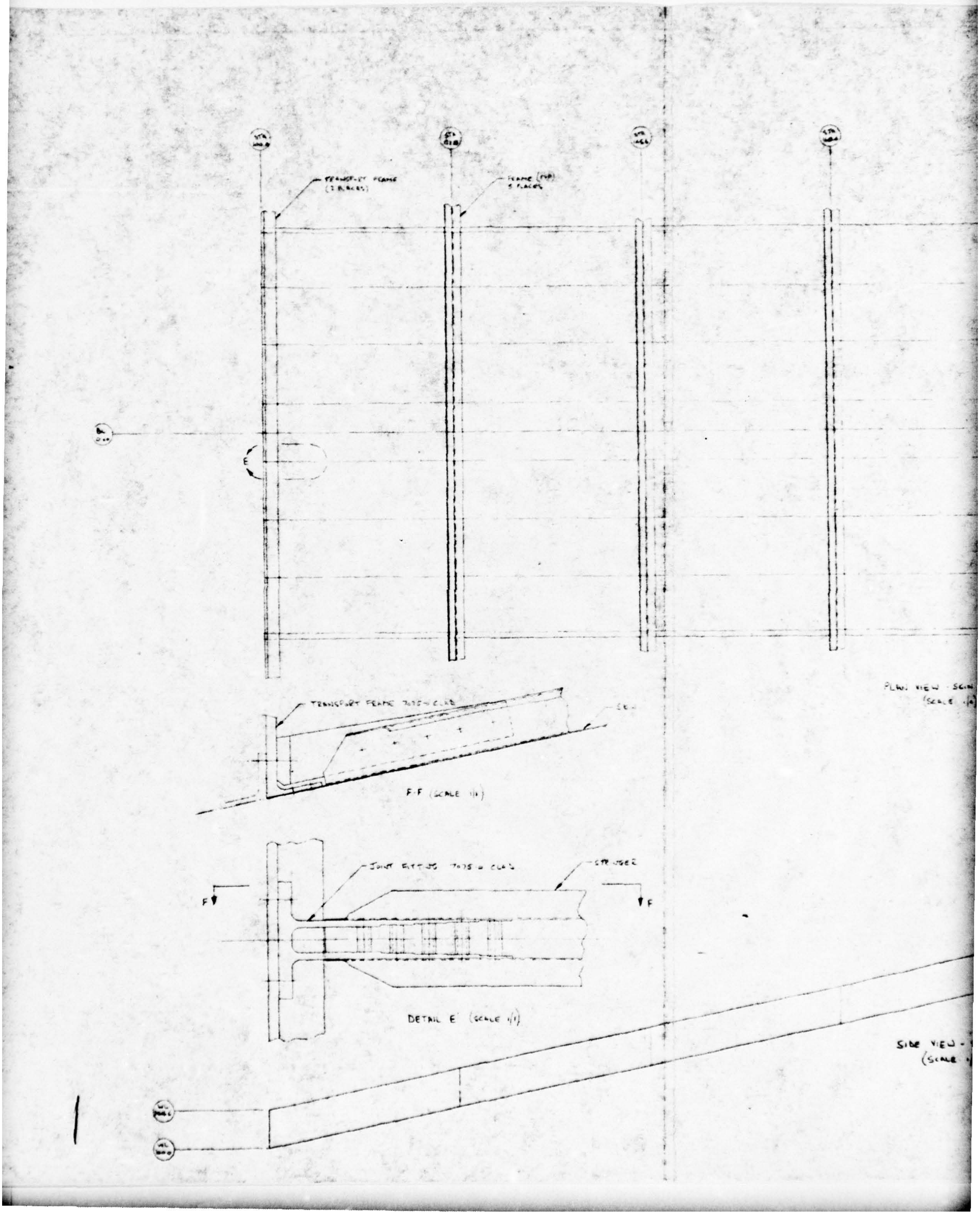
C. H. LAAKSO 2-20-75

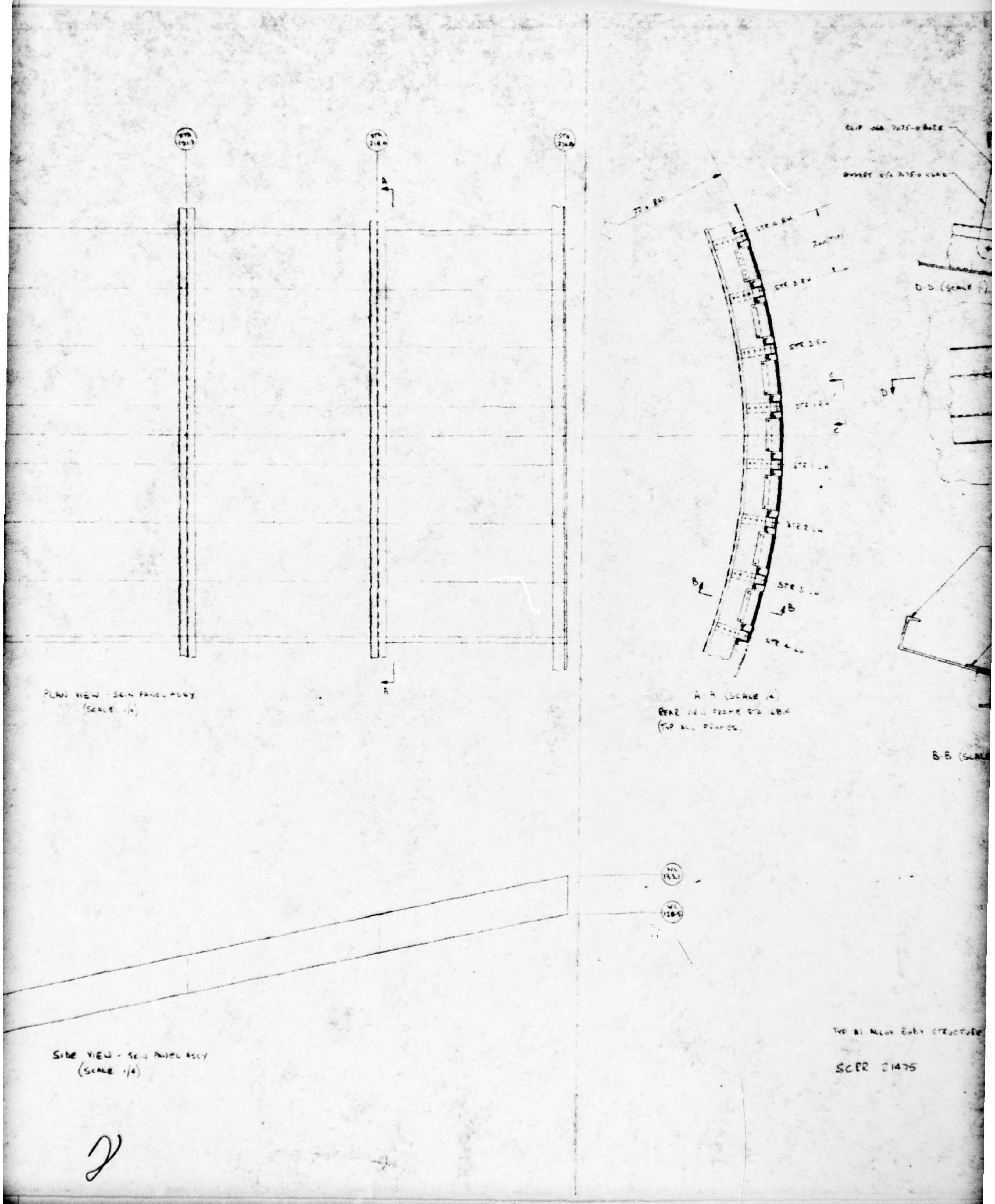
FORWARD LONGERON LOAD DISTRIBUTION PANEL SUBCOMPONENT No 1

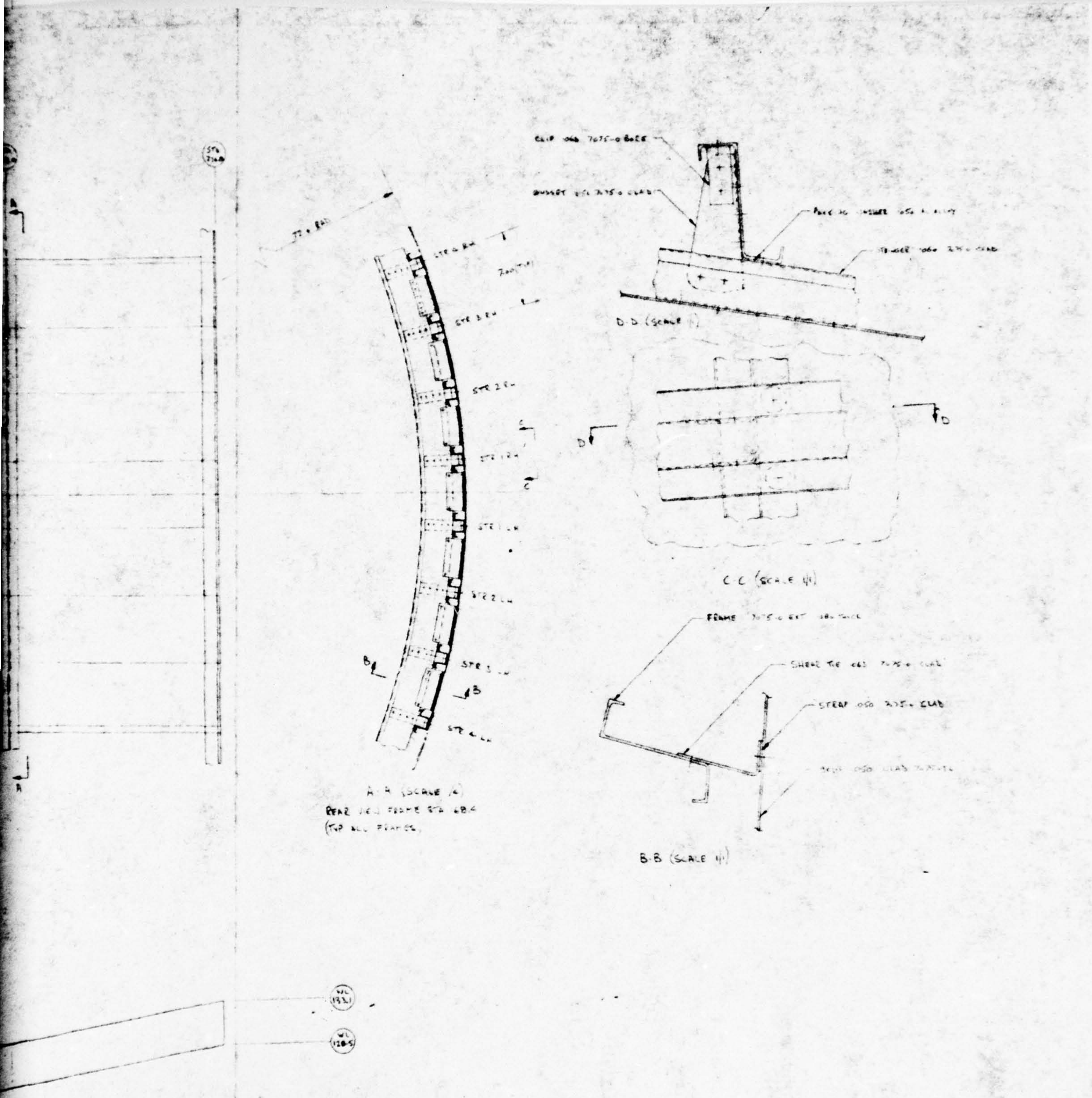
SK21775KO

A-8

3


TRANSPORT FRAME
FRAME (TYP)
E
F-F (SCALE 1/1)
JOINT FITTING 7075 CLAD
STRINGER
F
F
DETAIL E (SCALE 1/1)
PLAN VIEW
SIDE VIEW

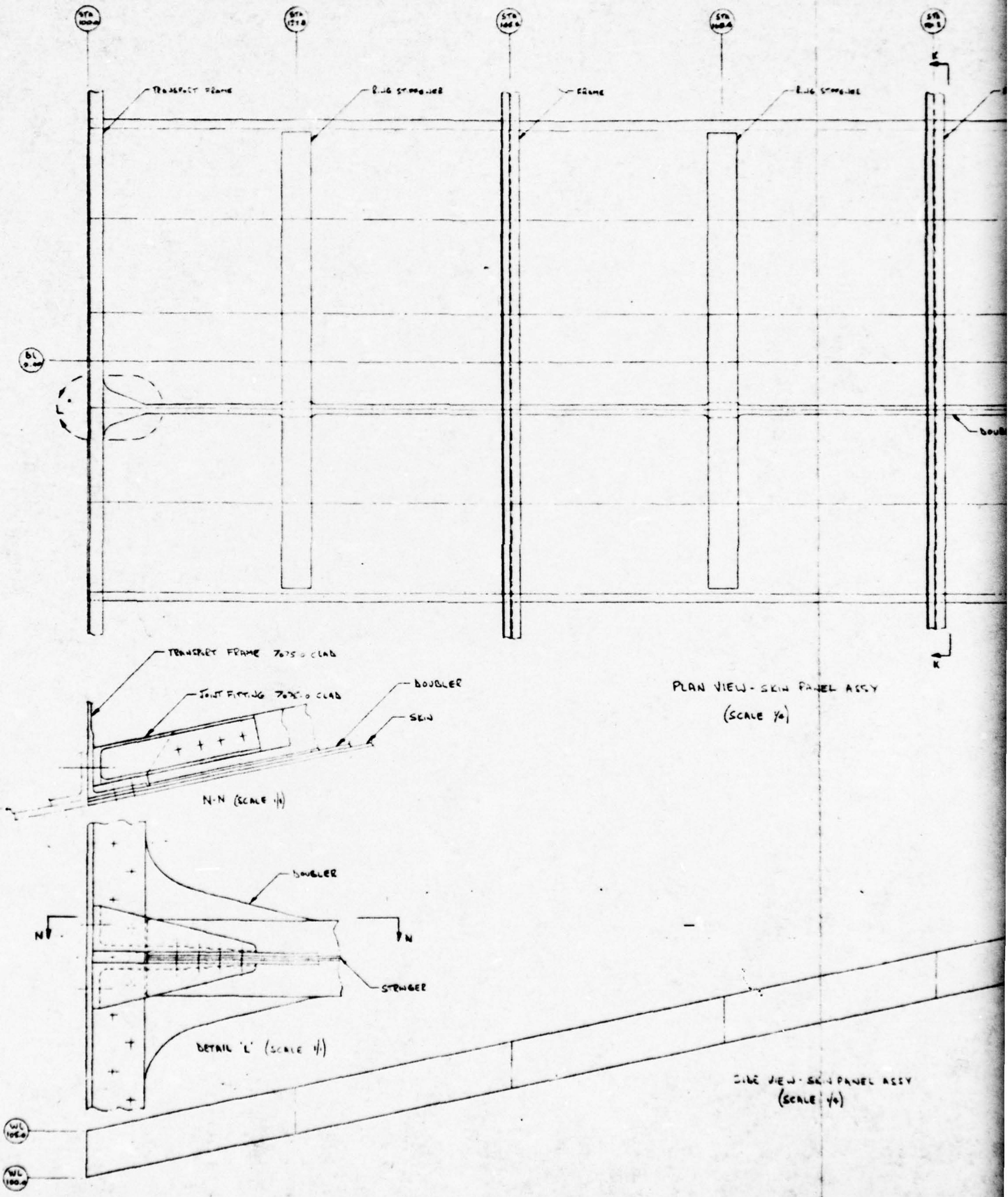

PLAN VIEW - SKIN PANEL ASSY
(SCALE 1/4)
A-A (SCALE 1/4)
D-D (SCALE
B-B (SCALE
SIDE VIEW - SKIN PANEL ASSY
(SCALE 1/4)
TYP AL ALLOY BODY STRUCTURE
SCPR 21475

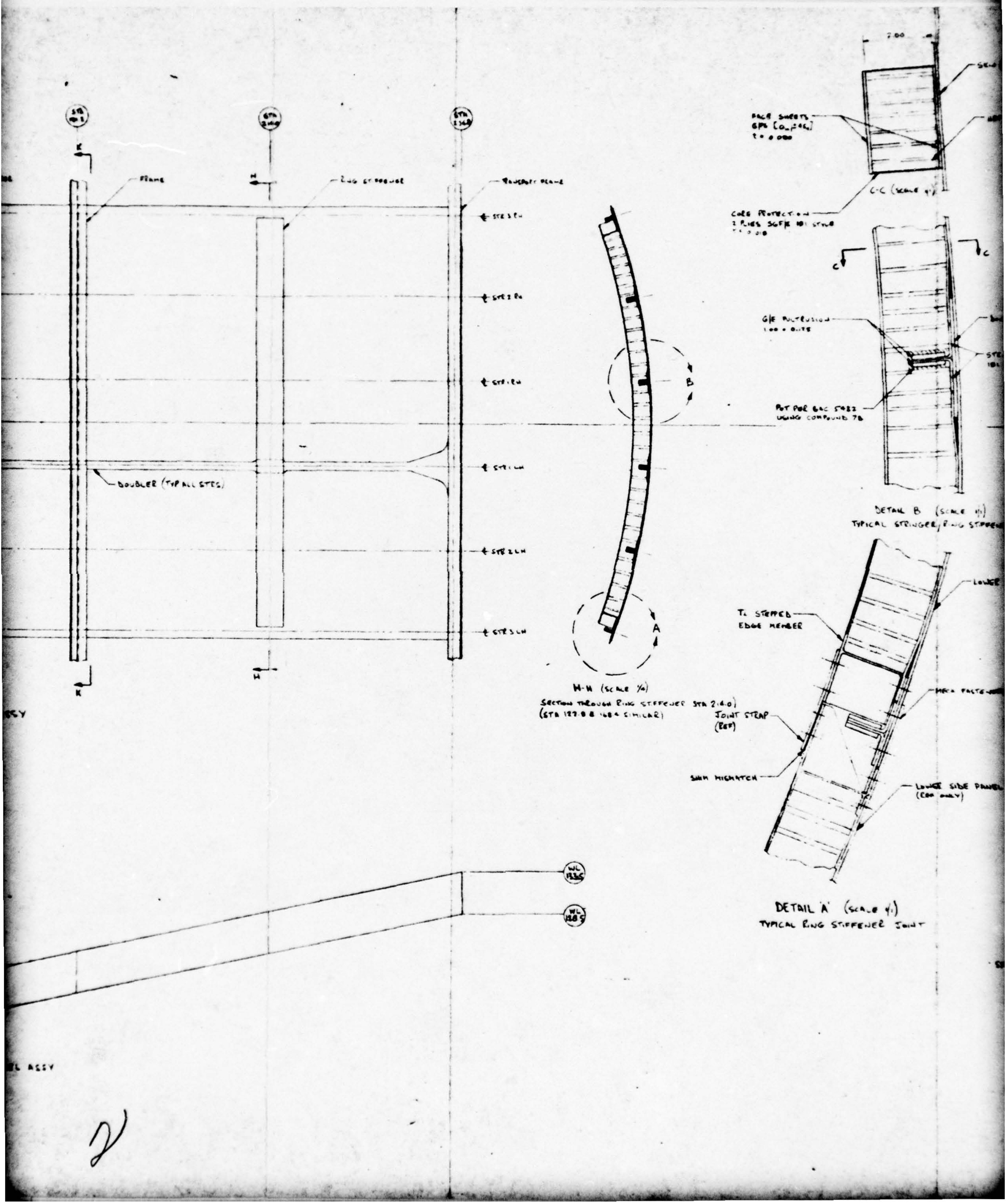


TYP AL ALLOY BODY STRUCTURE (UNPRESSURIZED)

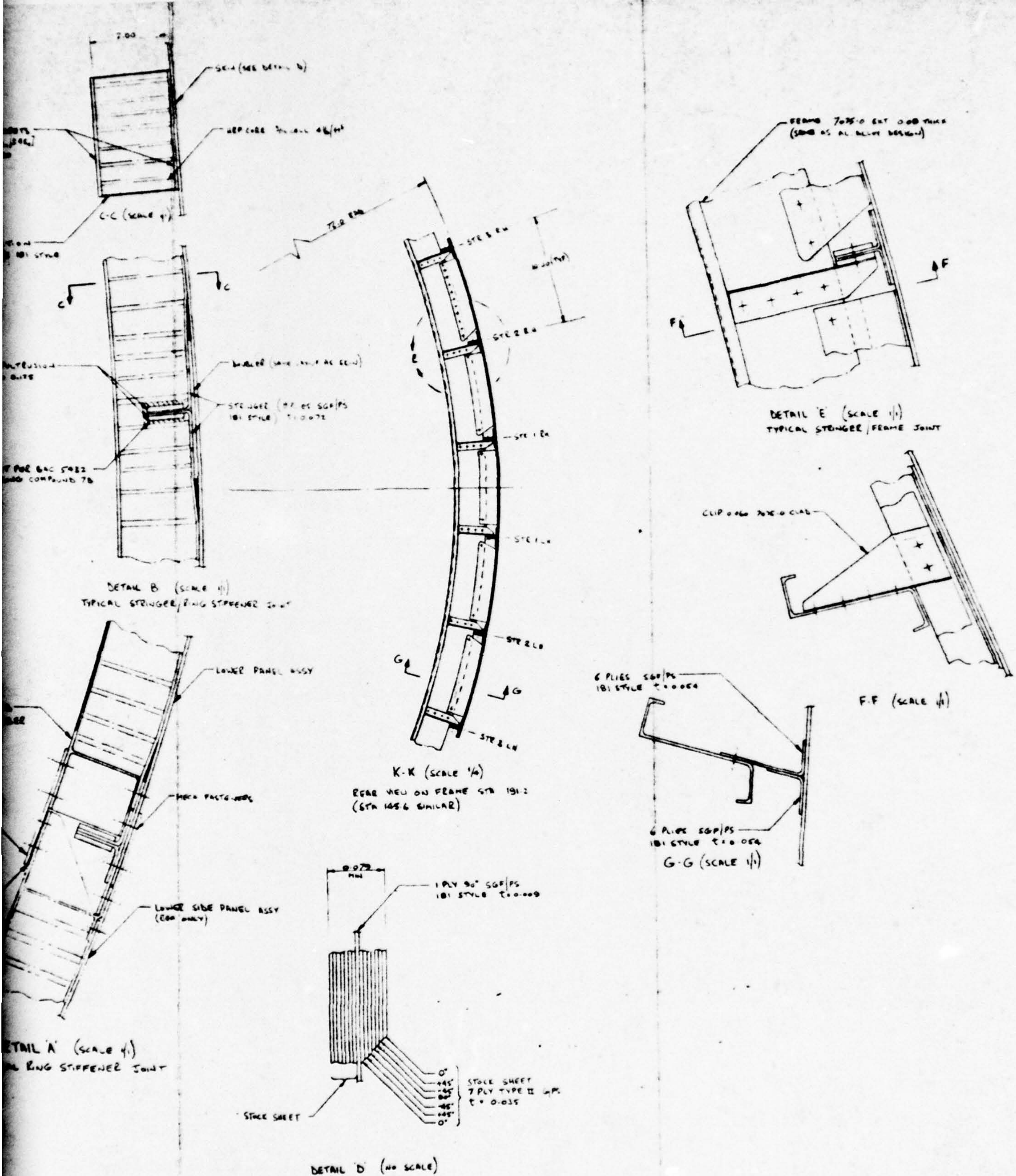
SCPR 21475

3

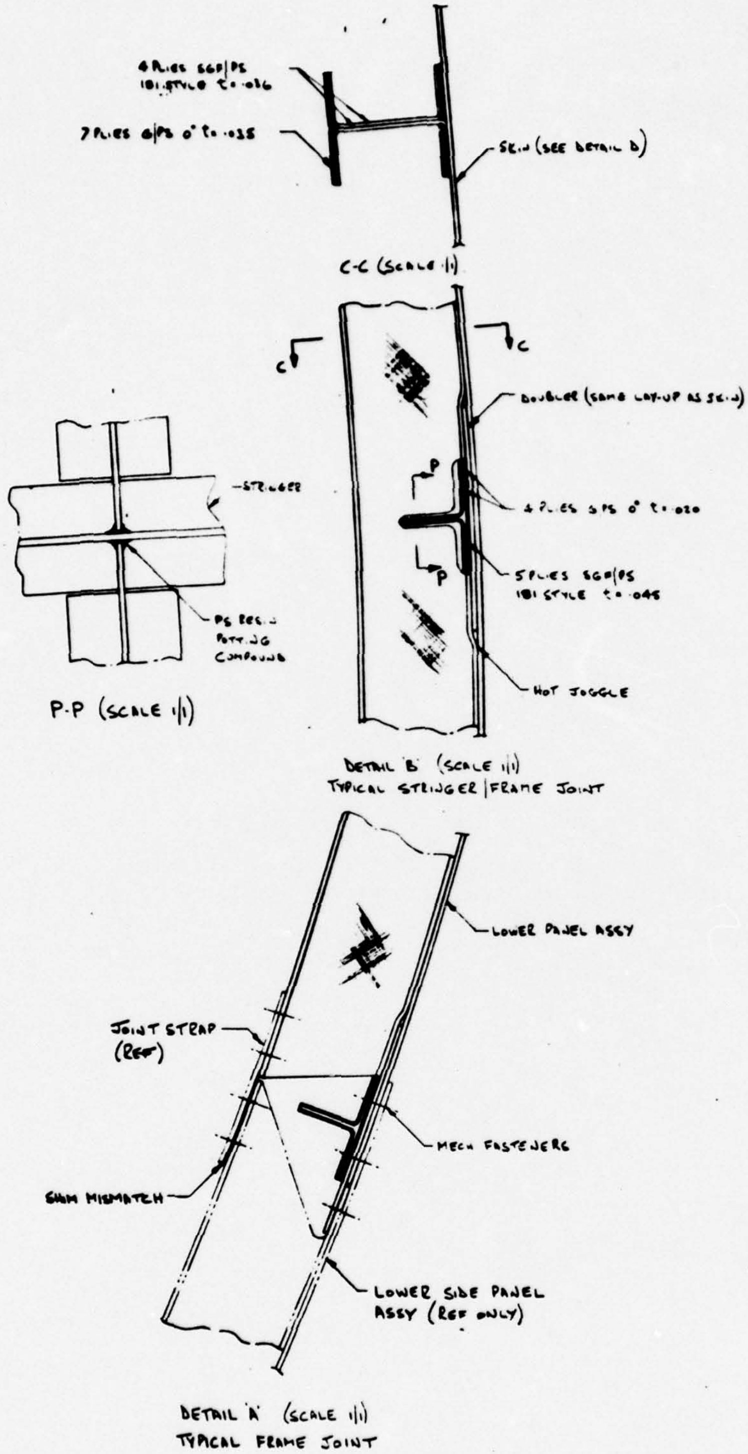

TRANSPORT FRAME
RING STIFFENER
FRAME
RING STIFFENER
DOUBLER
PLAN VIEW - SKIN PANEL ASSY
(SCALE 1/8)
TRANSPORT FRAME 7075-0 CLAD
JOINT FITTING 7075-0 CLAD
DOUBLER
SKIN
N-N (SCALE 1/1)
DOUBLER
STRINGER
DETAIL 'L' (SCALE 1/1)
SIDE VIEW - SKIN PANEL ASSY
(SCALE 1/8)



FRAME
RING STIFFENER
DOUBLER (TYP ALL STRS)
H-H (SCALE 1/4)
SECTION THROUGH RING STIFFENER
JOINT STRAP (REF)
SKIN MISMATCH
DETAIL B (SCALE 1/1)
DETAIL 'A' (SCALE 1/1)
TYPICAL RING STIFFENER JOINT
C-C (SCALE 1/1)
CORE PROTECTION
WL
STA



TYPICAL GRAPHITE REINFORCED POLYSULPHONE
BODY STRUCTURE (UNPRESSURISED)
SCRR 22075



I SECTION RING STIFFENER CONCEPT
(ALTERNATIVE TO H/C RING STIFFENER)

APPENDIX B
POST TEST ANALYSIS

The Gr/Ps Firebee center body section was successfully tested by NAVAIRDEVCEN to the (1) recovery and (2) 5g maneuver design limit conditions. Testing of the Gr/Ps section was accomplished in accordance with the procedures described in Section 10.0 (Component Test Program) of this report. Installation of the Gr/Ps components on the center body section of a XBQM-34E airframe is discussed in Section 3.0 (Demonstration Component Design). During testing, strain data was continuously recorded from strain gages located on the Gr/Ps components as shown in Figure B-1.

Both limit load test conditions were preceeded by application of ten 50% limit load cycles. No significant deviation in strain response was observed in the preliminary load cycling. The testing was then continued with application of five limit load cycles.

Figure B-2 summarizes the maximum corrected strain response recorded for the final cycle of two limit load test conditions. In general, no significant change was observed in the hardware and recorded strain data during the limit load cycles. Figures B-3 and B-4 are plots of selected strain gage data that illustrate that the reinforced skin panels did not buckle during testing.

In the 5g maneuver load tests, strains measured by gages 3, 4 and 5 on the upper skin panel changed significantly in the second load cycle to values that held essentially constant to the fifth load cycle.
Figure B-3 is a plot of the strain response shift that occurred at strain gage 4. Since no similar changes occurred in the more severe recovery condition tests, the strain response change was probably due to slip in the wing mount bolts or loading fixtures. In the second to fifth limit loadings, the strain data indicates that the skin panel responded with normal pre-buckling deformation having low resultant composite strains.

The structural analysis performed on the Gr/Ps components identified panel buckling as the most critical failure mode (see Figure 9-8, Margins-of-Safety Summary). Based on the component testing in which very low prebuckling strains developed, it is concluded that the Gr/Ps components would safely sustain the ultimate design load conditions.

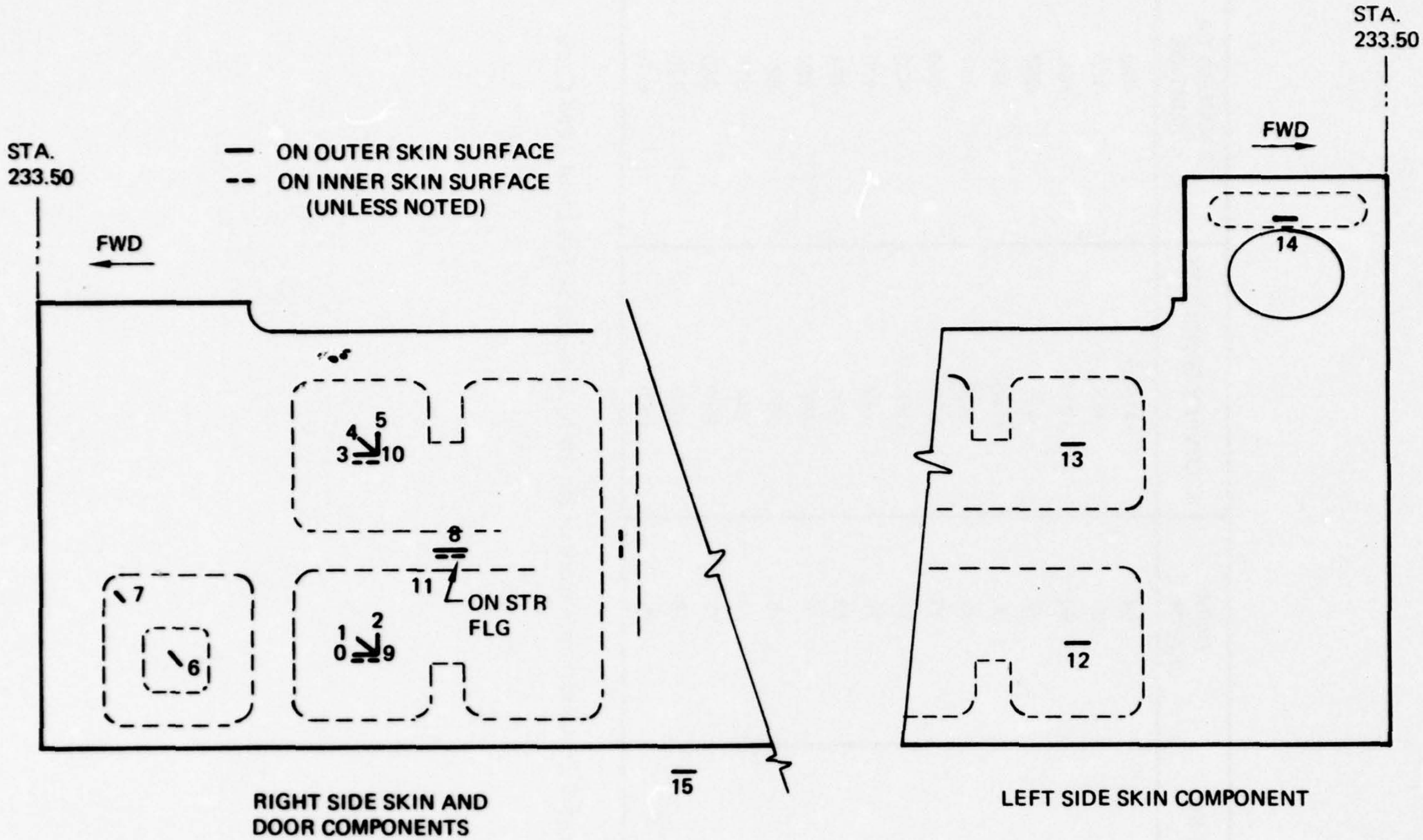


Figure B-1: Strain Gage Locations

GAGE NO.	DATA CHANNEL	RECOVERY CONDITION	5G MANEUVER CONDITION
0	64	-716 $\mu\epsilon$	-290
1	65	-118	-160
2	66	+267	+61
3	67	-498	-203
4	68	+383	-642
5	69	+429	-111
6	70	-682	-346
7	71	-1068	-432
8	72	-718	-420
9	73	-478	-204
10	74	-667	-298
11	75	-257	-52
12	76	-244	-326
13	77	-884	-340
14	78	-630	-170
15	79	+531	+216

Figure B-2: Maximum Recorded Strains (Corrected) In 5th Limit Load Cycle

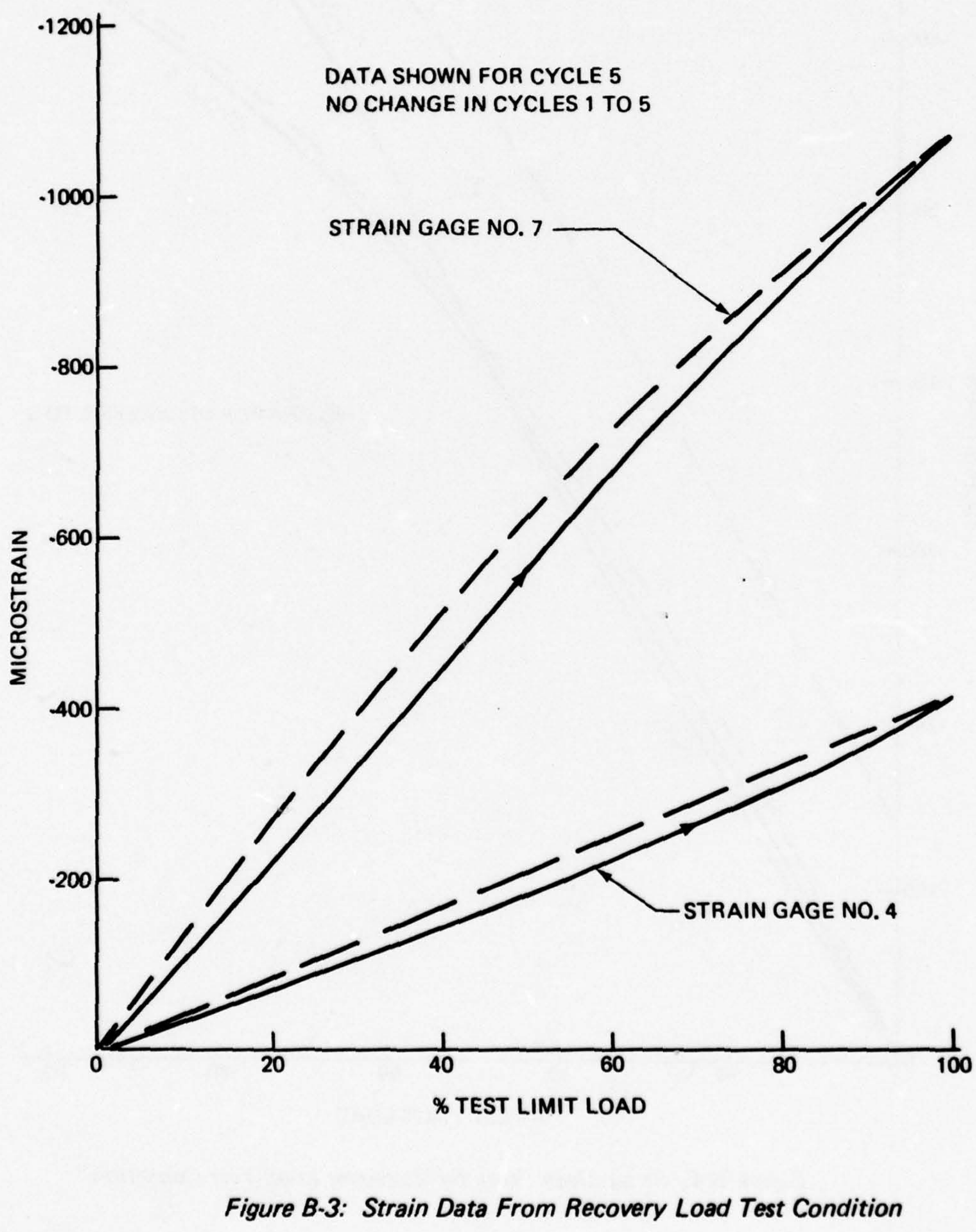


Figure B-3: Strain Data From Recovery Load Test Condition

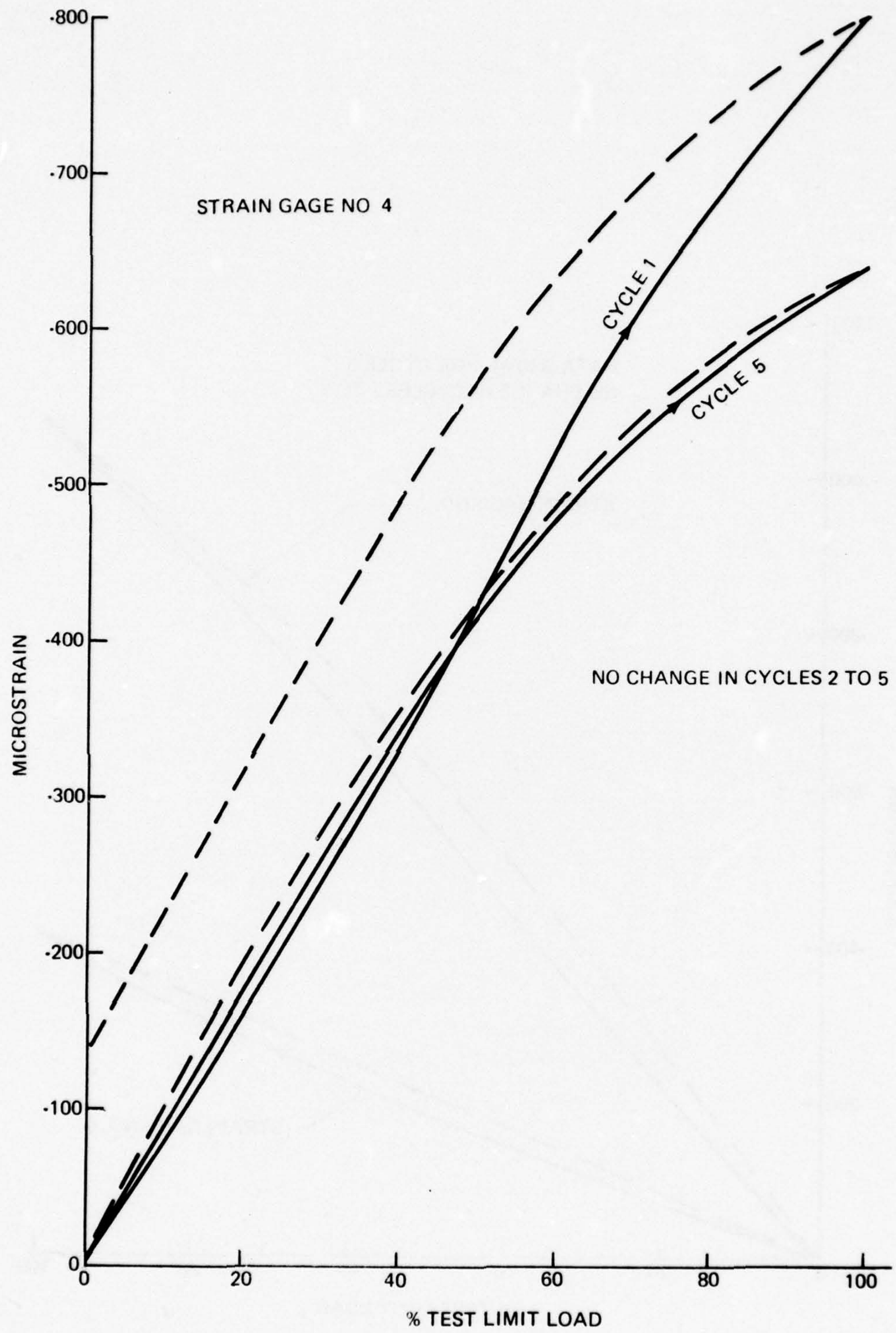


Figure B-4: Strain Data From 5g Maneuver Load Test Condition

APPENDIX C

Preliminary Material Specification

UNIDIRECTIONAL GRAPHITE PREIMPREGNATED
TAPE; THERMOPLASTIC MATRIX

Boeing Document D180-18236-4
June 1974

PRELIMINARY

Material Specification

UNIDIRECTIONAL GRAPHITE PREIMPREGNATED TAPE; THERMOPLASTIC MATRIX

JUNE 1974

Research & Engineering Division
THE BOEING AEROSPACE CO.
Seattle, Washington 98124

MATERIAL SPECIFICATION

TITLE: UNIDIRECTIONAL GRAPHITE PREIMPREGNATED TAPE; THERMOPLASTIC MATRIX

1.0 INTRODUCTION

1.1 Scope

This specification establishes the requirements for thermoplastic resin impregnated multi-filament collimated graphite tape of intermediate strength graphite reinforcement. The impregnated tape materials covered by this specification are intended for use in the manufacture of structural aircraft and missile components with service temperatures up to +250°F.

1.2 Classification

The material specified herein shall be classified in the following types:

Type I - Prepreg made from intermediate strength fiber tow material having a nominal fiber modulus of 30×10^6 psi.

Type II - Prepreg made from intermediate strength fiber yarn material having a nominal fiber modulus of 34×10^6 psi.

2.0 APPLICABLE DOCUMENTS

Except where specifically indicated, the issue of the following references in effect on the date of invitation for bid shall form a part of this specification to the extent specified herein.

a) MIL-STD-105 Sampling Procedures and Tables for Inspection by Attributes.

b) ASTM D792 Specific Gravity and Density of Plastics by Displacement, Test for

c) ASTM D790 Flexural Properties of Plastics, Methods of Testing.

d) ASTM D3039 Tensile Properties of Oriented Fiber Composites, Test for

e) ASTM D2344 Apparent Horizontal Shear Strength of Reinforced Plastics by Short-Beam Method, Test for,

f) MIL-P-46120A Plastic Molding & Extrusion Material, Polysulfone

3.0 REQUIREMENTS

3.1 Reinforcements

3.1.1 Quality

The reinforcements used for each prepreg type shall meet the requirements specified in Table I and the supplier shall so certify. The reinforcements shall be suitable for the production of quality finished parts and therefore shall not contain evidence of fuzzing, fiber deterioration, discontinuity, or foreign matter.

3.1.2 Finish

Fiber finish shall be complatible with the polysulfone matrix.

3.2 Matrix

The matrix material shall be a polysulfone resin per MIL-P-46120A Class I (Ref 2.0f). The resin shall contain no fillers or modifiers, and shall be free of foreign containments.

3.3 Prepreg Material

3.3.1 Physical Properties

The prepreg shall conform to the requirements specified in Table II when tested in accordance with the designated method.

3.3.2 Workmanship

a) The reinforcement shall be completely wetted with the resin matrix, both on the surface and throughout the fiber bundle.

b) The prepreg shall be free of foreign contaminants, wrinkles, twists, and visual fiber damage.

Table I: REINFORCEMENT PROPERTIES

FILAMENT PROPERTY	REQUIREMENT	
	TYPE I	TYPE II
Modulus of Elasticity, 10^6 psi	30 - 34	30 - 36
Tensile Strength, psi (minimum)	400,000	350,000
Filaments per Strand, (nominal)	10,000	3,000
Filament Diameter, Microns (nominal)	7.9	6.4
Yield, Yds. per Lb., (minimum)	480	2,600
Strand Twist per Foot, (maximum)	.2	.2

Table II: PHYSICAL PROPERTIES - PREIMPREGNATED MATERIAL

PROPERTY	REQUIREMENT		TEST METHOD
	TYPE I	TYPE II	
Resin Content, % by wt.	30 - 36	30 - 36	4.6.1
Volatiles, % by wt., max.	3.0	3.0	4.6.1
Tack	0	0	4.6.2
Width, inches*	3 ± .10	3 ± .10	-
Strands (Tows) per 3" width	12-13	56	-

* Unless Otherwise Stated on Purchase Order

c) Resin shall not flake or drop from the prepreg under routine handling.

d) Individual tows or strands shall be parallel to the tape centerline within ± 4 degrees along their entire length.

e) Open space between fibers or tows shall not exceed .010 inches wide, nor more than 10 inches long, cumulative total for each 10 square feet of prepreg for the Type I. Open space between strands shall not exceed .010 inches and strands shall be evenly spaced for the Type II.

f) Length and width of the material shall be specified on the purchase order.

g) Location of splices, etc. shall be clearly indicated on the separator paper.

3.3.3 Storage Life

The materials shall be capable of meeting all requirements of this specification for 1 year storage at + 77°F

3.4 Laminate Properties

Laminates, when fabricated per Section 6.3 from the preimpregnated materials, shall meet the physical requirements of Table III and the mechanical properties of Table IV.

3.5 Qualification

The graphite prepreg furnished to this specification shall be a product which has been tested and successfully passed all qualification tests specified herein. All material purchased to this specification shall have been qualified for listing on this specification's qualified products list (QPL) at the time set for opening of bids.

Table III: PHYSICAL PROPERTIES OF LAMINATE

PROPERTY	REQUIREMENT		TEST METHOD
	TYPE I	TYPE II	
Specific Gravity, g/cc	1.53 (Min)	1.53 (Min)	2.0b
Fiber Volume Content, %	56 - 62	56 - 62	4.6.3
Void Content, %	3, (Max)	3, (Max)	4.6.3
Thickness per Ply, inches	.007 - .008	.005 - .006	4.6.4

Table IV: MECHANICAL PROPERTIES OF LAMINATE

PROPERTY	REQUIREMENT [1] TYPE I	TYPE II	TEST METHOD
Flexural Strength, psi			
$+70^{\circ}F$	175,000	175,000	4.6.5
$+250^{\circ}F$	130,000	130,000	4.6.5
Flexural Modulus, 10^6 psi			
$+70^{\circ}F$	13	13	4.6.5
$+250^{\circ}F$	12	12	4.6.5
Tensile Strength, psi			
$+70^{\circ}F$	180,000	180,000	4.6.6
Tensile Modulus, 10^6 psi			
$+70^{\circ}F$	16	16	4.6.6
Interlaminal Shear Strength, psi			
$+70^{\circ}F$	10,000 *	10,000 *	4.6.7
$+250^{\circ}F$	6,000 *	6,000 *	4.6.7

[1] Minimum Average of 5 Specimens, Normalized, Except as Noted, to 60% Fiber Volume. Elevated Temperature Tests Require 0.1 Hr. Soak at Temperature. All Specimens Nominally 0.10" Thick for Flexural and ILS property determinations and 0.04" thick for tension property determinations.

* Values Not Normalized.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 Responsibility for Inspection

The vendor shall supply all samples and shall be responsible for performing all required tests. Results of all tests shall be reported to the purchaser as required by 6.2. Purchaser reserves the right to perform such confirmatory testing as deemed necessary to assure that material conforms to the requirements of this specification.

4.2 Classification of Tests

4.2.1 Qualification Testing

Qualification testing shall include all examinations and tests required by this specification.

4.2.2 Quality Conformance

Quality conformance tests shall include the tests specified in Table V.

4.3 Supplier Quality Control

The Quality Conformance Tests indicated in Table IV shall be performed by the supplier * on each production lot of preimpregnated material.

Suppliers shall furnish actual test data showing conformance to this specification and shall identify such data with the applicable specification revision letter in effect. The supplier shall include a copy of prepreg test data for each batch of prepreg offered. All of the identifying information of 5.1 shall be included in the prepreg test data.

The supplier shall maintain for a period of one year records of the certified properties of each lot of raw material used in production of prepreg purchased to this specification.

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* Suppliers shall either have the facilities required to test in accordance with this specification or shall utilize the services of an outside laboratory, subject to approval of the purchaser's Quality Assurance organization.

Table V: CLASSIFICATION OF TESTS

Tests	Requirement	Qualification Test	Quality Conformance Test	Test Method
PREPREG TESTS				
Nongraphite Content	3.3.1	X	X	4.6.1
Volatiles	3.3.1	X	X	4.6.1
Tack	3.3.1	X	X	4.6.2
Visual Examination	3.3.2	X	X	3.3.2
Storage Life	3.3.3	X		3.3.3
LAMINATE PHYSICAL TESTS				
Specific Gravity	3.4	X	X	2.b
Fiber Volume	3.4	X	X	4.6.3
Void Content	3.4	X	X	4.6.3
Thickness per ply	3.4	X	X	4.6.4
LAMINATE MECHANICAL TESTS				
Tensile Strength & Modulus	3.4	X		4.6.6
Flexure Strength & Modulus	3.4	X	X	4.6.5
Interlaminar Shear Strength	3.4	X	X	4.6.7

The properties shall include as a minimum the reinforcement properties of Table I, and confirmation that the matrix resin conforms to the requirements of MIL-P-46120A Class I, (2.0f).

Certified raw material properties shall be made available to purchaser Quality Control on request.

4.4 Purchase Quality Control

a) Purchase Quality Control shall review supplier test data and perform any additional inspection or testing necessary to assure that the production material meets all the requirements of this specification.

b) Purchase Quality Control shall perform the following tests as a minimum on each production lot of material:

(1) *Visual Examination per 3.3.2.*

(2) Prepreg physical tests per Table II.

(3) Any additional inspections or testing deemed necessary to assure that the preimpregnated material meets all the requirements of this specification.

4.5 Sampling

Sampling Schedule: Prepreg property measurements shall be in accordance with Single Sampling for Normal Inspection, General Inspection Level II, with an Acceptable Quality Level (AQL) of 1.5 specified in MIL-STD-105, (ref. 2a) as shown in Table II. Test specimens shall be taken at random throughout the lot, except that there shall be no splices in test material. Mechanical properties on laminates, per Tables III and IV, shall be determined once per lot. The frequency of sampling for laminate properties shall be not less than 4 determinations for each test, unless otherwise specified.

Table V: SAMPLING SCHEDULES

Number of Spools Per Lot	Number of Spools From Which Samples Are to Be Taken	Accept	Reject
1	1	0	1
2 - 8	2	0	1
9 - 15	3	0	1
16 - 25	5	0	1
26 - 50	8	0	1
51 - 90	13	0	1
91 - 150	20	1	2
151 - 280	32	1	2
281 - 500	50	2	3

4.6 Material Test Methods

4.6.1 Resin Content and Volatile Content

(a) Cut three samples two inches square, from different areas of the prepreg, combine, and weigh to the nearest milligram (W_1)

(b) Dry the samples for 30 minutes at +250 - 275°F and re-weigh to the nearest milligram (W_2)

(c) Place all three samples in a beaker containing 100 ml of methylene chloride and allow to stand approximately 5 minutes with slight agitation.

(d) Decant the solvent and repeat Step (c) two more times, each time taking extreme care not to loose any fibers in the decanting process.

(e) Dry the fibers for 60 minutes at 250°F - 275°F.

(f) Weigh fibers to the nearest milligram (W_3)

(g) Calculate:

$$\% \text{ Volatile Content} = \frac{W_1 - W_2}{W_1} \times 100$$

Where: W_1 = Original Sample Weight

W_2 = Dried Sample Weight

$$\text{Resin Content } \% = \frac{W_2 - W_3}{W_2} \times 100$$

Where: W_2 = Sample Weight Prior to Extraction

W_3 = Weight of Extracted and Dried Fibers.

4.6.2 Tack

Prepreg shall be considered to have zero tack when folded upon itself and the plies do not adhere together when finger pressure is applied.

If the plies adhere, the material shall be rejected.

4.6.3 Fiber Volume and Void Content

(a) At least 10 determinations shall be made for qualification testing and at least 4 determinations for acceptance testing.

(b) Cut 1-inch square samples from as many separate panels made for Laminate Mechanical Properties (Table IV) as possible (or taken from tested specimens), and weigh each sample to the nearest milligram, (W_c)

(c) Place each sample in a beaker containing 100 ml of methylene chloride, cover, and allow to stand 24 hours at room temperature.

(d) Decant the solvent from the fibers. Thoroughly wash the fibers with fresh methylene chloride taking care during the washing and decanting operations not to loose any fibers.

(e) Dry the fibers for 1 hour at $250^{\circ}F - 275^{\circ}F$, then allow to stand at $+72^{\circ}F \pm 5^{\circ}F$ and $50 \pm 10\%$ relative humidity for 2 hours minimum.

(f) Weigh the fibers to the nearest milligram, W_f.

(g) Calculate fiber volume (V_f):

$$\% \text{ Fiber Volume} = V_f = \frac{W_f}{\rho_f} \times \frac{\rho_c}{W_c} \times 100$$

Where W_f = Weight of Fiber

W_c = Original Weight of Composite Sample

ρ_f = Density of Fiber (From Supplier)

ρ_c = Density of Composite per 2.0 b

(h) Calculate Void Content (V_o):

$$\% \text{ Void Content} = V_o = 100 - \left(V_f + \frac{W_r}{\rho_r} \frac{\rho_c}{W_c} \right)$$

Where W_r = $W_c - W_f$

ρ_r = Density of Resin (From Supplier)

4.6.4 Ply Thickness

Thickness per ply shall be determined by measurement of the panels prepared for tensile and flexural test or standard control panel using a ball-tipped micrometer. The reported ply thickness shall be the average of at least 10 determinations on each panel measured

$$\text{Ply Thickness} = \frac{\text{Laminate Thickness}}{\text{No. of Plys in Laminate}}$$

4.6.5 Flexural Strength and Modulus

Flexural strength and modulus shall be determined in accordance with Reference 2.0c, Method 1, Procedure A using a span-to-depth ratio of 32:1 and a loading fixture having 1/4 inch diameter points.

4.6.6 Tensile Strength and Modulus

Tensile strength and modulus shall be determined in accordance with Reference 2.0d.

4.6.7 Interlaminar Shear Strength

Interlaminar shear strength determinations shall be conducted in accordance with Reference 2.0e at a span-to-depth ratio of 4:1.

5.0 PREPARATION FOR DELIVERY

5.1 Material Identification

Place the following information on a label on each prepreg container.

a) Vendor Material Designation and/or Identification Number
b) Lot Number _____
c) Boeing Material Specification _____. Type _____
d) Boeing Purchase Order No. _____
e) Quantity _____ Width _____
f) Manufacturer ______________________
g) Date of Impregnation ______
h) Resin Content ______
i) Volatile Content ______
j) Inspected by ________________
k) Expiration Date _______ (12 months from impregnation date)
l) Graphite Lot No. _______
m) Resin Lot No. _______

5.2 Packaging

5.2.1 Packaging Spool

Unless otherwise specified, tape material shall be wound on spools not less than 7.5 inches in diameter and interleaved with nonadherent paper. Winding shall be uniform and shall provide for proper unreeling. Tape ends shall be secured.

5.2.2 Package Sealing

Unless otherwise specified, each spool of material shall be sealed in an individual bag of suitable nonadherent material to prevent contamination and penetration of moisture.

5.3 Exterior Packaging

5.3.1 Packing

The protected spools of material shall be packed in an exterior shipping container identified with a package number and capable of protecting the materials adequately during transit, and meeting the minimum requirements of carrier rules and regulations applicable to the mode of transportation.

5.3.2 Marking of Exterior Package

Each exterior shipping container shall be legibly marked with the following information in such a manner that the markings shall not smear or be obliterated during normal handling or use.

TAPE; GRAPHITE, THERMOPLASTIC RESIN IMPREGNATED TOW (YARN)

PURCHASE ORDER NUMBER ______________________________

MANUFACTURER'S MATERIAL DESIGNATION ________________

LOT AND PACKAGE NUMBERS _________________________

QUANTITY ____

EXPIRATION DATE ________________

6.0 NOTES

6.1 Intended Use

The material described herein is intended for use in making structural composites for use temperature to +250°F where a high ratio of stiffness-to-weight or strength-to-weight is required. The various types described reflect the range of commercially available graphite fiber.

6.2 Qualification

(a) Direct all requests for qualification to a representative of the Materiel Department of The Boeing Company who will specify data and samples desired for qualification purposes.

(b) Test data supplied in duplicate must give individual test values obtained (and not use the word "conforms") showing the material meets all the requirements of this specification for type submitted. The test facility (supplier or test laboratory) used in determination of the data shall be identified.

(c) The qualification sample shall consist of one representative production sample of the particular type under which the supplier elects fo qualify.

(d) Qualification requires successful completion of all requirements of this specification.

(e) After review of supplier data and Boeing tests by the applicable Engineering Staff-Materials and Processes Unit, the supplier will be advised as to whether or not the material has been qualified.

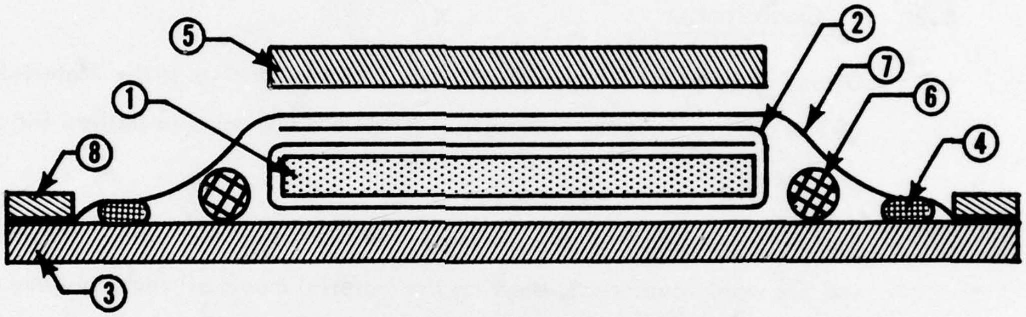
(f) Vendor shall establish the control factors of processing which were used to produce material meeting all requirements of this specification. These shall constitute the approved procedures and shall be used for manufacturing production material. If necessary to make any change in control factors of processing which could affect quality or properties of the material, vendor shall submit for reapproval a statement of the revised procedures and, when requested, sample material. No product material incorporating the revised procedures shall be shipped prior to receipt of approval.

6.3 Laminating Procedure

Laminates for Qualification and Quality Assurance tests shall be fabricated as follows:

a) Lay up the required number of plies* of prepreg into a unidirectional laminate and vacuum bag as shown in Figure 1.

FIGURE 1 BAGGING SYSTEM



(1) Graphite/Polysulfone Laminate

(2) Parting film, 5 mil Kapton (6.5b) coated with Frekote 33 (6.5c) on part side, Note: 1 1/2 times around part prevents side wash of fibers.

(3) Base Plate

(4) Bag Sealant, high temp., (6.5e).

(5) Pressure plate, shown as used in press. (Place inside bag and cover with 181 glass cloth for autoclave use.)

(6) Bleeder cloth, press position (6.5f)

(7) Bag film, 2 mil Kapton (6.5a)

(8) Clamping frame -used to hold down edge of bag

b) Cure, using an autoclave or press, according to the following schedule and maintain greater than 20" Hg vacuum throughout entire cure.

(1) Heat at convenient rate to 650°F

(2) Apply 200 psi

(3) Hold for 10-30 mins. depending on part heat-up rate (i.e.,autoclave generally 10 mins. hold time and preheated press about 30 mins.)

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* Number of prepreg plies will vary depending on the type of prepreg (Type I or II) and the laminate thickness (0.040" or 0.10").

(4) Cool under pressure to 200°F at a rate of 3°F/min (maximum).

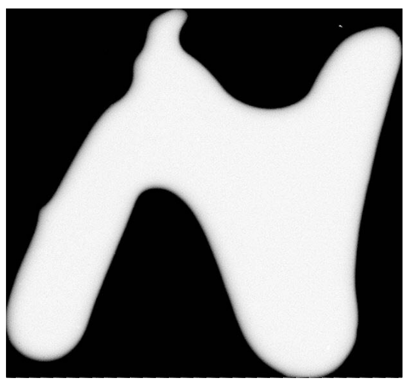
6.4 Lot

A lot is defined as all the material produced in a single production run, made from the same batches of raw materials, under the same fixed conditions, and offered for delivery at one time.

6.5 Materials

a) Bag film, 2 mil thick, Kapton film, type H, E.I. duPont deNemours & Co.
b) Parting film, 5 mil thick, Kapton film, type H, E.I. duPont deNemours & Co.
c) Mold Release, Frekote #33, Frekote, Inc., Boca Raton, Fla. 33432
d) Methylene Chloride, reagent grade.
e) Bag sealant, high temp (650°F), suggest freshly milled and extruded silicone rubber stock (uncatalysed) S.W.S. 7220, 20 duro.
f) Bleeder cloth, any suitable glass fabric.

END



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DEVELOPMENT OF A LOW-COST GRAPHITE REINFORCED COMPOSITE PRIMARY--ETC(U)
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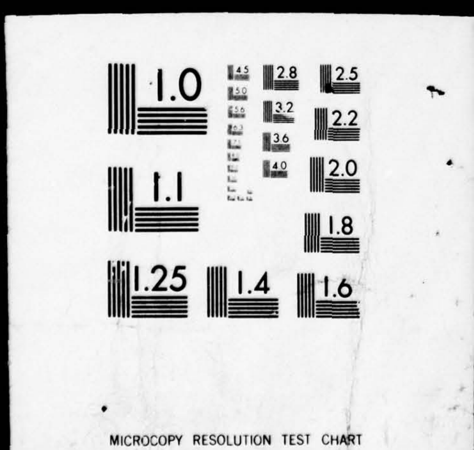

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ERRATA

Reference: NADC Contract N62269-74-C-0368 - Development of a Low-Cost Graphite Reinforced Composite Primary Structural Component

Several errors were inadvertently incorporated in the subject report and should be noted.

Page 6 - Next to last sentence of last paragraph should read "Because of increased structural efficiency of the Gr/Ps composite, ring stiffener segments are not needed to replace the metal ring sections cut away at stations 250.06 and 266.62."

Page 13 - 2nd line, "isotopic" should read "isotropic."

Page 22 - 2nd line from bottom, "memeer" should read "member."

Page 42 - 3rd line, 2nd paragraph, delete "the equipment used for the holographic inspection was a Holosonics brand, Model 100 Through Transmission Acoustic Holography System."

Page 68 - Last line, Figure 94 should read Figure 9-4.

ERRATA